

# UNCLASSIFIED

AD NUMBER
AD477910
NEW LIMITATION CHANGE
TO Approved for public release, distribution unlimited
FROM Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; NOV 1964. Other requests shall be referred to Army Transportation Research Command, Fort Eustis, VA.
AUTHORITY
USAAMRDL ltr 23 Jun 1971

THIS PAGE IS UNCLASSIFIED

1  
me

U. S. A R M Y

TRANSPORTATION RESEARCH COMMAND

FORT EUSTIS, VIRGINIA

472910

6

PRELIMINARY FLIGHT TEST DATA.

XH-51A RIGID ROTOR HIGH SPEED FLIGHT PROGRAM.

9

INTERIM REPORT, NO. 7

11

NOV 1964,

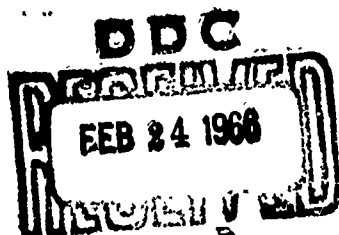
12

36p.

AD No.

DDC FILE COPY

3



042 200



The information contained herein has been reproduced to disseminate to Government and industry, as rapidly as possible, current data vital to Army rotary-wing progress and objectives. In this respect, it is emphasized that the data, although measured flight data, are preliminary; therefore, the contents of this document are subject to revision.

ACCESSION FOR	
CFSTI	WHITE SECTION
DDO	DIFF. SECTION
UNANNOUNCED	
JUSTIFICATION	
BY	
DISTRIBUTION/AVAILABILITY CODES	
BYST.	AVAIL. BY SPECIAL

2

## SUMMARY

This report summarizes the flight test results of the Phase III four-bladed rotor testing on the XH-51A "rigid rotor" helicopter. The object of Phase III was to evaluate the four-blade rotor in comparison with the three-blade rotor. Ground tests commenced on 13 July 1964 with the first flight on 21 July 1964. The phase was terminated on 3 October 1964 after 49 flights and 14.9 flight hours.

## Results and Discussion

### General

Two problems were apparent as a result of early flights, blade stresses and vibration level - both were high. At this stage, the transmission was in soft suspension in pitch only with a spring rate of 14,400 pounds/inch; vertically and laterally the mount was solid. The fuselage vibration problem was dominantly 4P; it was considered that this resulted from the second and third harmonics of blade natural flapping frequency which were close to 3P and 5P. The 3P and 5P rotating inputs resulted in a 4P shaft oscillation which was transmitted to the fuselage. The fuselage responds rather readily to frequencies in this range. The transmission suspension was varied incrementally to tune it to a point of minimum transmissibility; this approach had not been completed at the close of Phase III. Additionally, experimental moves were made towards cutting down the input by reducing the blade natural flapping frequencies. The configuration at the termination of the phase, resulting from these two approaches, incorporated a 13-pound weight at station 6.0 on each of the four blades and the transmission soft mounted in pitch, roll, and vertically. Blade stresses were reduced to within the infinite life level and vibration, although unacceptable from a production viewpoint, was also considerably reduced.

Apart from the vibration level, no flying qualities problems were encountered. A marked increase in the Phase III flight envelope was achieved, 2.5 g being demonstrated from 0 to 120 knots CAS and a speed of 180 knots IAS flown in a shallow dive.

Phase III employed four blades of the same design used on the three-blade rotor; the hub blade attachment geometry was modified compared with that employed in the Phase II flying. The only significant problem apparent at the close of Phase III was fuselage vibration, none of the stability problems encountered with the three-blade rotor existed and stress levels, subject to fatigue test confirmation, indicate infinite life for the hub. With the reduction of the vibration level to within normal limits, a substantial speed envelope expansion should be possible. The phase is considered a considerable success in that a marked increase in

flight envelope was demonstrated, no stability problems existed, and no problems considered insurmountable were encountered. The stability characteristics confirmed the advisability of the choice of the four-blade rotor for the compound configuration.

#### Configuration

Four-blade rotor - The blade design was that employed during Phase II, the three-blade rotor; however, the blade hub attachment was modified. The cone angle built into the hub was increased from 2.8 degrees to 3.2 degrees.

Main rotor gearbox suspension varied as described.

Tailplane angle of incidence  $-5\frac{1}{2}^{\circ}$  through test 271 and  $-3^{\circ}$  thereafter.

Blade weights 13 pounds each, fitted to each main rotor blade at station 6.0 for test 352 and remained on thereafter.

Four-arm gyro 7.3 slug ft<sup>2</sup>.

Arm incidence 30 degrees.

#### Structures

Structural loads recorded during the program included main rotor hub and blades, gyro control arms, main rotor pitch link, tail rotor and tailplane loads. An incremental approach was employed during the tests, flight records being examined prior to further envelope expansion.

Figures 1 and 2 compare Phase II and III flight envelopes. At a C.G. of 1.5" forward, the envelope is substantially 2.5 to 0.2 g up to 120 knots CAS with reducing normal acceleration cut to 165 knots CAS. The aft envelope was opened up in only three or four flights and no specific attempt was made to exceed the Phase II envelope. While it is felt that there would be little point in extending the hover beyond the 0.15 to 2.7 g demonstrated, no structural, performance, or stability limits were encountered in forward flight.

#### Rotor Stresses

A review of all structural data indicates that hub station 7.0 is the critical fatigue section of the rotor. It consists of three steel laminations bonded and bolted together. Assuming a stress concentration of 3, the endurance limit stress is 26,000 psi. The strain calibrations were affected in terms of bending moment rather than stress because the bending moment curve along the span of hub and blade is predictable. The conversion of bending moment to stress at station 7.0 is as follows:

Flapwise bending      Station 6.0 moment x 1.42 =  
station 7.0 stress

Chordwise bending      Station 6.0 moment x 1.152 =  
station 7.0 stress

Figure 3 shows that during the initial flights, the chordwise bending stress was 40 percent lower than that in the three-blade, but the flapping stress was up by 80 to 90 percent. At this stage, the vibration was very high and a series of changes in transmission suspension springs was initiated to reduce both vibration and stress. The pitch spring rate was varied first; the range covered was 6,400 pounds/inch to solid while both vertical and lateral remained solid. From the structural loads and vibration results 11,000 pounds/inch was selected as the pitch spring rate to be held constant during vertical spring variations. The vertical range covered was from solid down to 4,000 pounds/inch and 6,000 pounds/inch was selected from the results.

An analysis of the flapwise bending at station 6.0 during forward flight and flare showed considerable 3P and 5P content superimposed on the 1P. Note, 1P is main rotor rotational frequency, normally 5.9P cps (100T), 3P is 3 times rotor frequency, etc. The second and third harmonic blade natural flap bending frequencies were slightly below the 3P and 5P forcing frequencies. Tests conducted at 95 percent, 100 percent, and 105 percent illustrated the diminishing response of the blade as the forcing frequency was increased and separated from the natural. The 3P and 5P bending moments were reduced by approximately 45 and 55 percent respectively. The reduction of the blade natural frequencies had a similar effect; at 100 knots, the 3P bending moment was reduced from 4,090 inch pounds to 2,400 inch pounds and the 5P from 1,960 inch pounds to 740 inch pounds. In reducing these moments, the weights reduced the 3P and 5P driving forces which produced the 4P pitching and rolling moment into the fuselage. The vibration benefited to a minimum degree.

To reduce the 4P moment further, a transmission lateral spring was introduced with a rate of 19,000 pounds/inch; it did reduce the cabin vibration to a small degree, but did not affect the structural loads.

Records showed a 1P flapping component increasing with speed at high speed and producing a nose down moment and indicating that a reduction of tailplane negative incidence would be of value in reducing the total bending moment. The tailplane incidence was changed from  $-5\frac{1}{2}^{\circ}$  to  $-3^{\circ}$  for test 228. The change in slope of the flapwise bending moment curve between figures 4 and 5 at high speed is attributable to this change.

Figures 6 and 7 illustrate the affect of normal acceleration on the flapwise bending moment at station 6.0 at two centers of gravity. The average moment increased towards up flapping with increasing load factor due to lift on the rotor blade. Down flapping was recorded at 1.0 g due to the fact that the blade line was below the cone angle built into the hub. Zero moment was recorded when the blade at this station lined up with the hub cone angle. The smaller built-in cone angle on the three-blade rotor and the 30 percent greater lift per blade resulted in the different intercept shown in the graphs. The Phase III flapping cyclic stresses, figure 7, at the aft C.G. were of the order of 10 percent lower than those at a mid-C.G. on Phase II. At the forward C.G., figure 6, they were about 30 percent lower than the Phase II mid-C.G. Chordwise average and cyclic moments in maneuver were 50 percent lower than with the three-blade rotor except at high load factors where the reduction in average moment was about 10 percent. The cyclic flapwise and chordwise stresses at station 6.0 are the maximum that occurred during the maneuvers and do not necessarily coincide with the maximum load factor.

The spanwise bending moment distribution in high speed level flight, descent, and in maneuver are shown in figures 10 through 15.

#### Vibration

Vibration level in the cabin was measured for speeds up to 132 knots CAS in level flight and to 165 knots CAS in a descent. Tri-axis (vertical, fore and aft, lateral) measurements were made on the cabin floor at the pilot's seat. Vibration data in the 3 axes is plotted versus airspeed (CAS), figure 16.

The analysis of vibration for the various configuration was carried out in conjunction with the structural loads, because they were in most cases a function of each other. The comparison with the three-blade data is obvious, but it must be pointed out that the three-blade data represents only the soft cabin configuration. The high vibration level remaining at the termination of Phase III was the only problem of any significance and was the factor which limited the speeds attainable under this contract.

#### Stability

No adverse stability characteristics were recorded at any time throughout the Phase III flying. There was a tendency for the static

longitudinal stability to become neutral at high speed and some cyclic cross-coupling existed. The correction of the cross-coupling would have improved the stability which could have been made further positive by aerodynamic shaping of the gyro arms. For the purposes of the program, neither the stability nor the coupling warranted corrective action.

The cyclic control to trim is shown in figures 17 and 19 in terms of control position and maximum blade incidence. Figure 18 shows the results of constant collective static stick fixed and free longitudinal stability measurements.

Figures 20 and 21 illustrate the longitudinal and lateral control power and 22 compares Phase II and Phase III results. The lateral control power was increased by 14 percent to 12.0 degrees/second-inch and in both phases and was independent of speed. Longitudinally, the 20 to 25 percent increase apparent at the lower airspeeds decreased with speed to become zero at about 110 knots IAS. The data was obtained from longitudinal step inputs in the hover and at 70 and 110 knots IAS. The time histories showed compliance with MIL-H-8501A in that the angular velocity was in the proper direction within 0.2 seconds of the control displacement and the point of inflection of normal acceleration occurred within 1.0 second of the control displacement.

The stick force per g plots in figures 23, 24, and 25 illustrate the greatest single improvement of the four-blade rotor relative to the three-blade rotor. No pitch up, nor any tendency for the stick force/g to be other than positive was experienced during Phase III. The stick force/g has improved 150 to 200 percent, providing a high degree of confidence with regard to the anticipated outcome of maneuver tests at high airspeeds. An envelope of the bank angles flown during these maneuvers is presented in figure 26.

A number of entries and autorotations were affected in the range 80 to 120 knots CAS; the characteristics were normal and perfectly acceptable. The load factor to hold rotor speed presented in figure 27 stems from tests conducted to assist in the definition of the technique to be used following failure of the rotor engine on the compound helicopter.

#### Performance

Performance data presented in figure 29 is not representative of the four-bladed clean helicopter configuration in that at the time the tests were conducted, the blade weights were installed, and they represent a significant and unknown drag increment. The hover data presented in figure 28 is clean four-blade data.



DATA APPENDIX  
INDEX

<u>Figure</u>	<u>Title</u>
1	Flight Envelope - C.G. 1.46" Forward . . . . .
2	Flight Envelope - C.G. 2.15" Aft . . . . .
3	Moment at Hub Station 6.0 - Initial Phase III Data . . . . .
4	Moment at Hub Station 6.0 - Transmission Sus- pension Modified Blade Weights Incorporated . .
5	Moment at Hub Station 6.0 - Final Phase III Configuration - Tailplane -3° . . . . .
6	Flap Bending Moment at Station 6.0 vs. Load Fac- tor C.G. 1.46" Forward - Final Phase III Con- figuration . . . . .
7	Flap Bending Moment at Station 6.0 vs. Load Factor - C.G. 2.15" Aft - Final Phase III Con- figuration . . . . .
8	Chord Bending Moment at Hub Station 6.0 vs. Load Factor - C.G. 1.46" Forward - Final Phase III Configuration . . . . .
9	Chord Bending Moment at Hub Station 6.0 vs. Load Factor - C.G. 2.15" Aft - Final Phase III Configuration . . . . .
10	Flap Bending Moment at hub Station 6.0 vs. Span Level Flight - Final Phase III Configuration .

<u>Figure</u>	<u>Title</u>
11	Chord Bending Moment at Station 6.0 vs. Span Level Flight - Final Phase III Configuration
12	Flap Bending Moment at Hub Station 6.0 vs. Span in Dive - Final Phase III Configuration
13	Chord Bending Moment at station 6.0 vs. Span in Dive - Final Phase III Configuration .
14	Flap Bending Moment at Hub Station 6.0 vs. Span in Maneuver - Final Phase III Configu- ration . . . . .
15	Chord Bending Moment at Station 6.0 vs. Span in Maneuver - Final Phase III Configuration.
16	Cabin Vibration Level - Final Phase III Configuration . . . . .
17	Control to Trim in Level Flight . . . . .
18	Constant Collective Static Longitudinal Sta- bility . . . . .
19	Blade Incidence to Trim in Level Flight . .
20	Longitudinal Control Power . . . . .
21	Lateral Control Power . . . . .
22	Control Power Comparison - 3-blade and 4- blade Rotors . . . . .
23	Maneuvering Stability - 4-Blade System . . .
24	Maneuvering Stability - 3-Blade Rotor . . .
25	Maneuvering Stability - 3-Blade Rotor . . .
26	Angle of Bank Velocity Envelope . . . . .
27	Load Factor - Rotor Speed in Autorotation .

Figure

Title

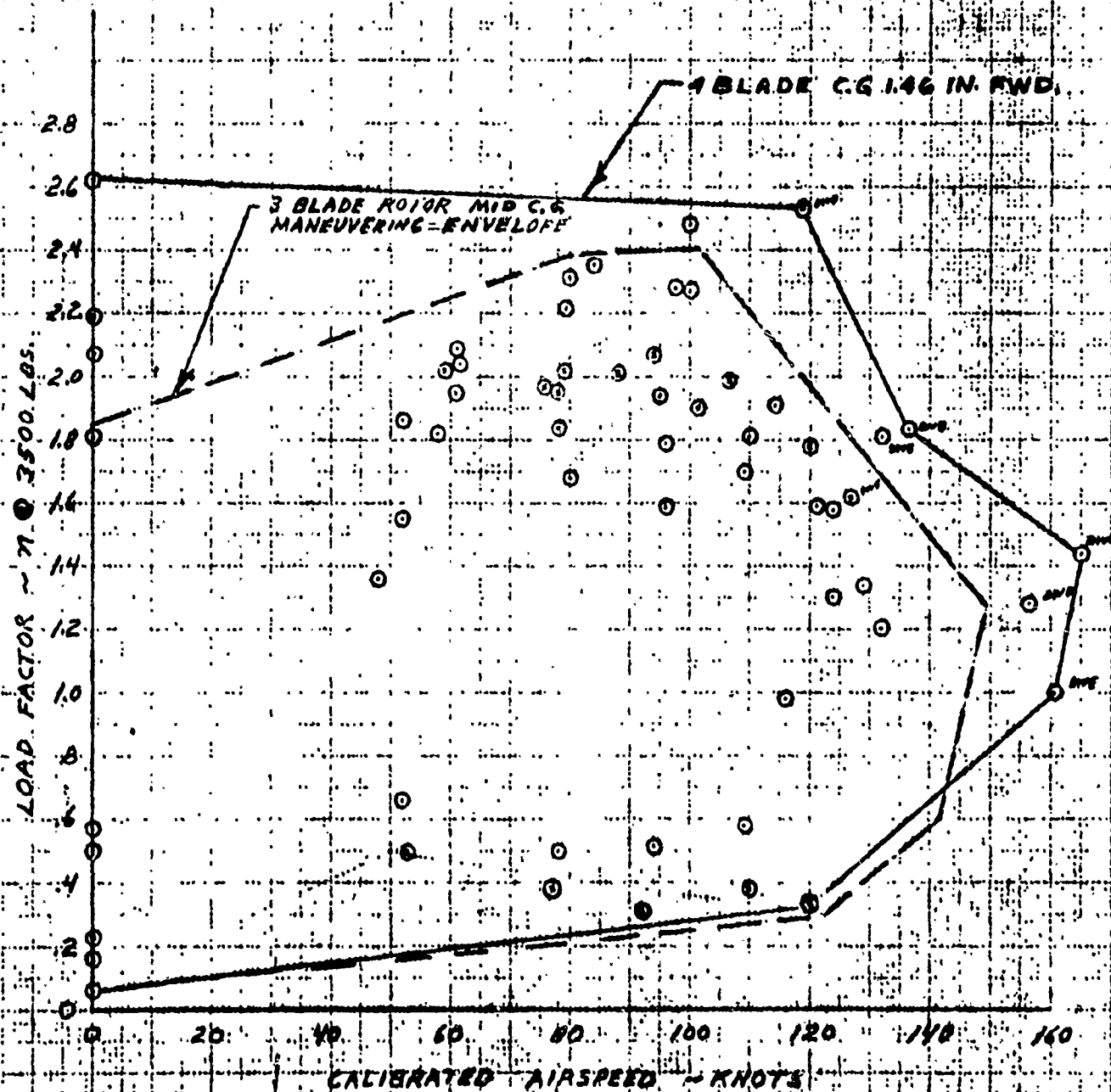
28	Hover Performance . . . . .
29	Forward Flight Performance . . . . .

# V-n DIAGRAM

GROSS WEIGHT - 3500 LBS

C.G. 1.46 IN. FWD

4 BLADE ROTOR



FORM 8870A

FLIGHT ENVELOPE C.G. 1.46" FWD.

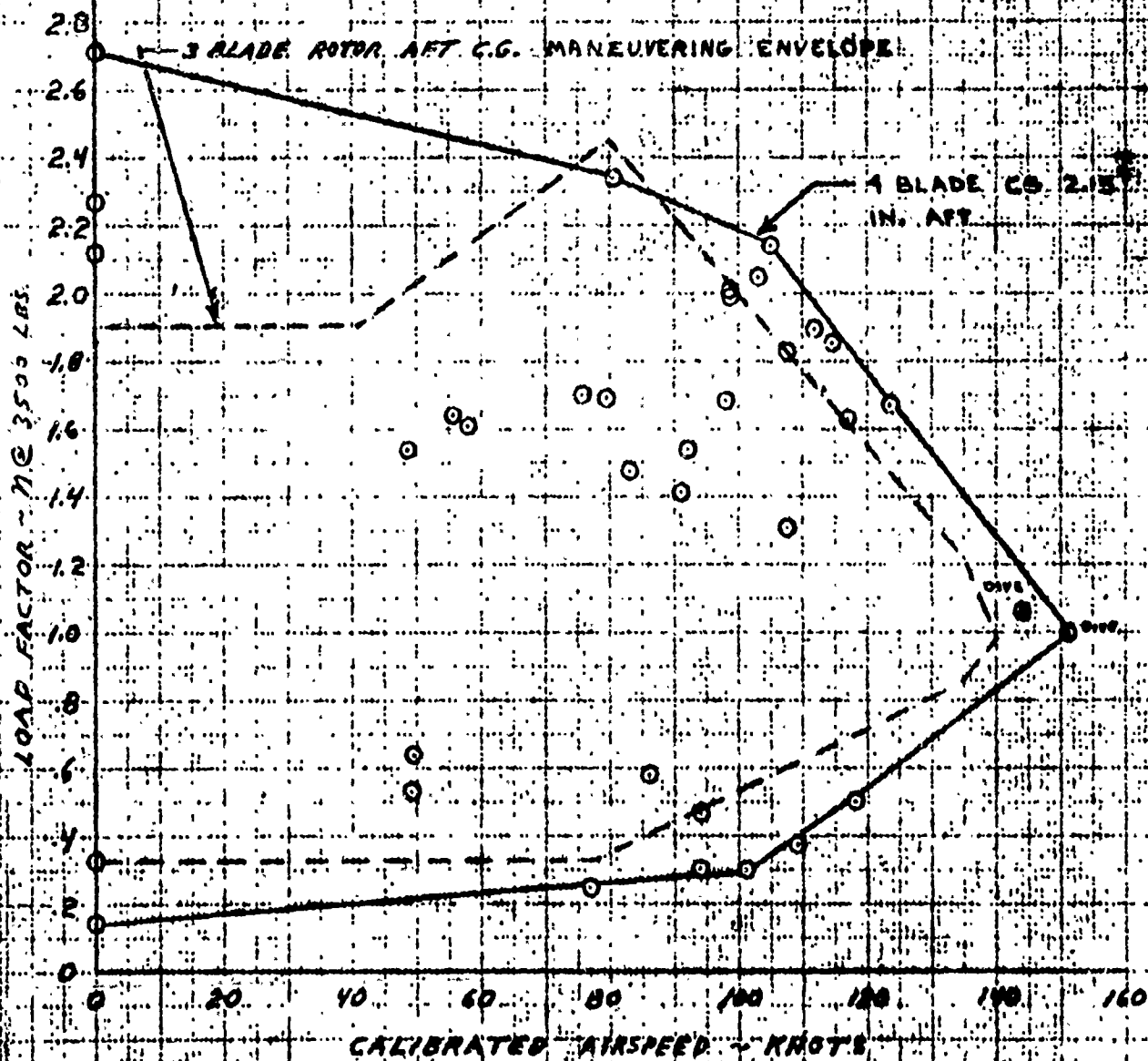
S.F.61.  
FIG. 1

# V-71 DIAGRAM

GROSS WEIGHT: 3500 LBS

C.G. 2.15" AFT

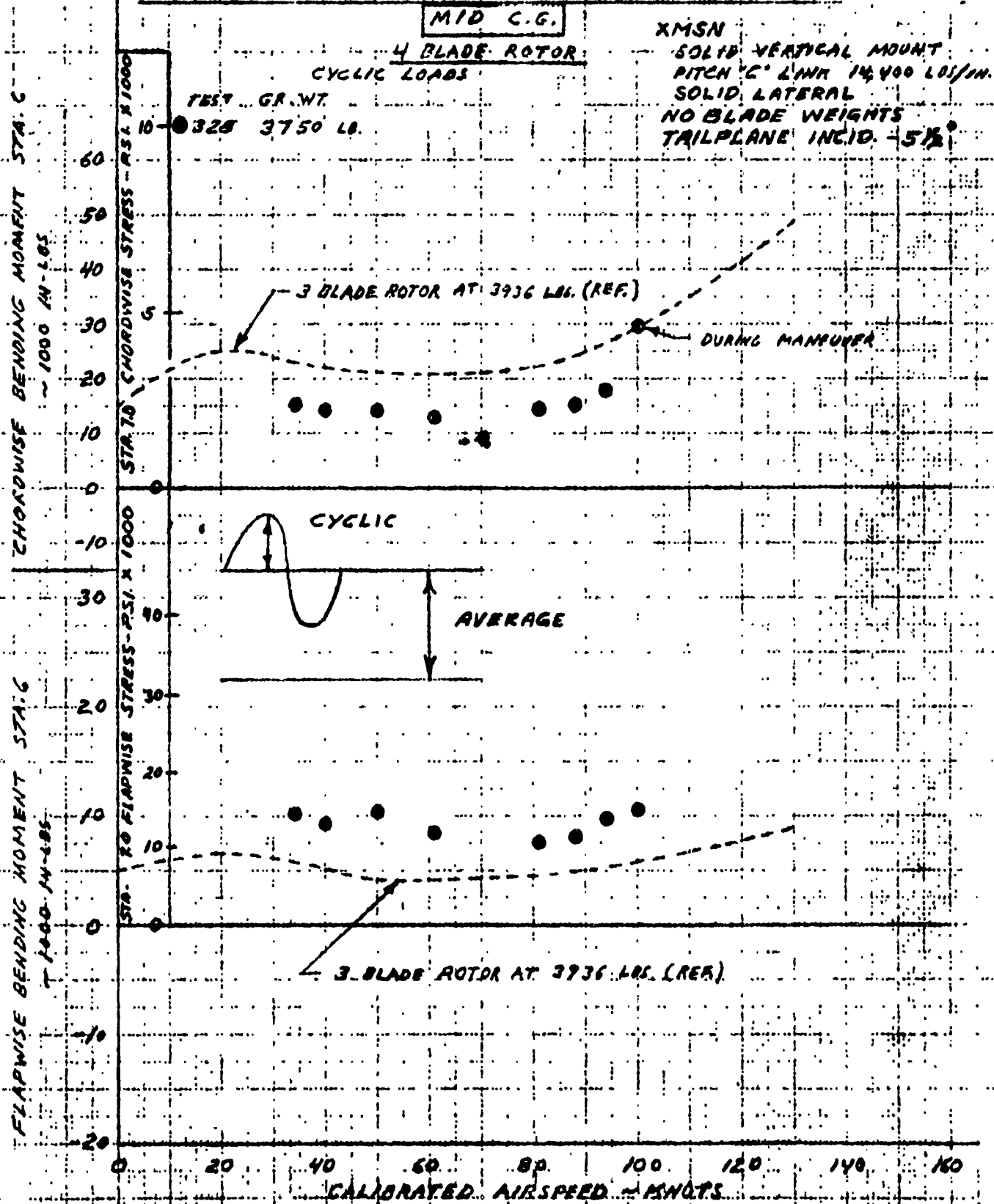
4 BLADE ROTOR



FLIGHT ENVELOPE C.G. 2.15" AFT

SF 60.1  
FIG 2

# MAIN ROTOR BLADE LOADS V. CALIBRATED AIRSPEED



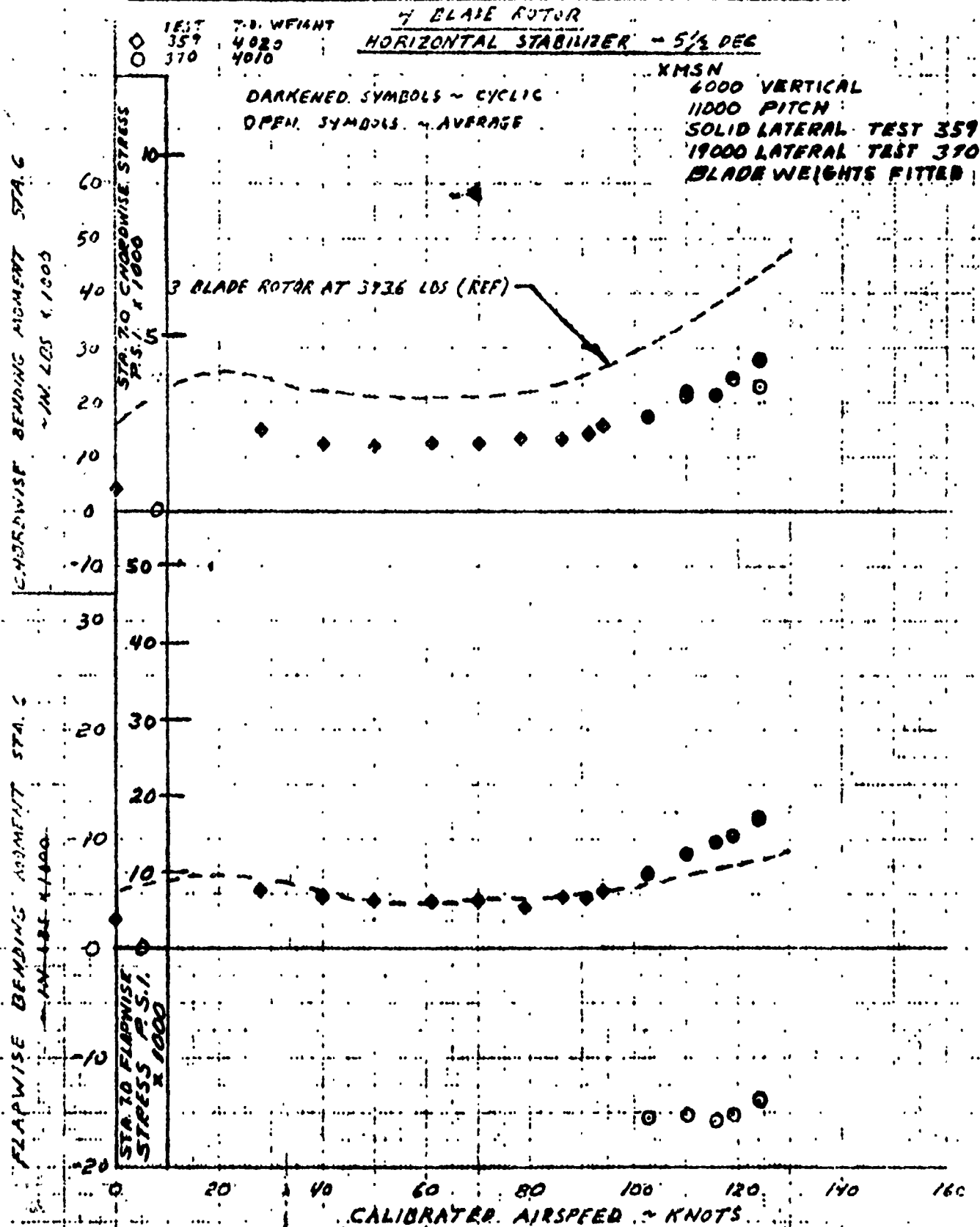
FORM 8878A

MOMENT AT HUB STA. 6.0

INITIAL PHASE II DATA

EX 26  
FIG 3

# MAIN ROTOR BLADE LOADS VS. CALIBRATED AIRSPEED



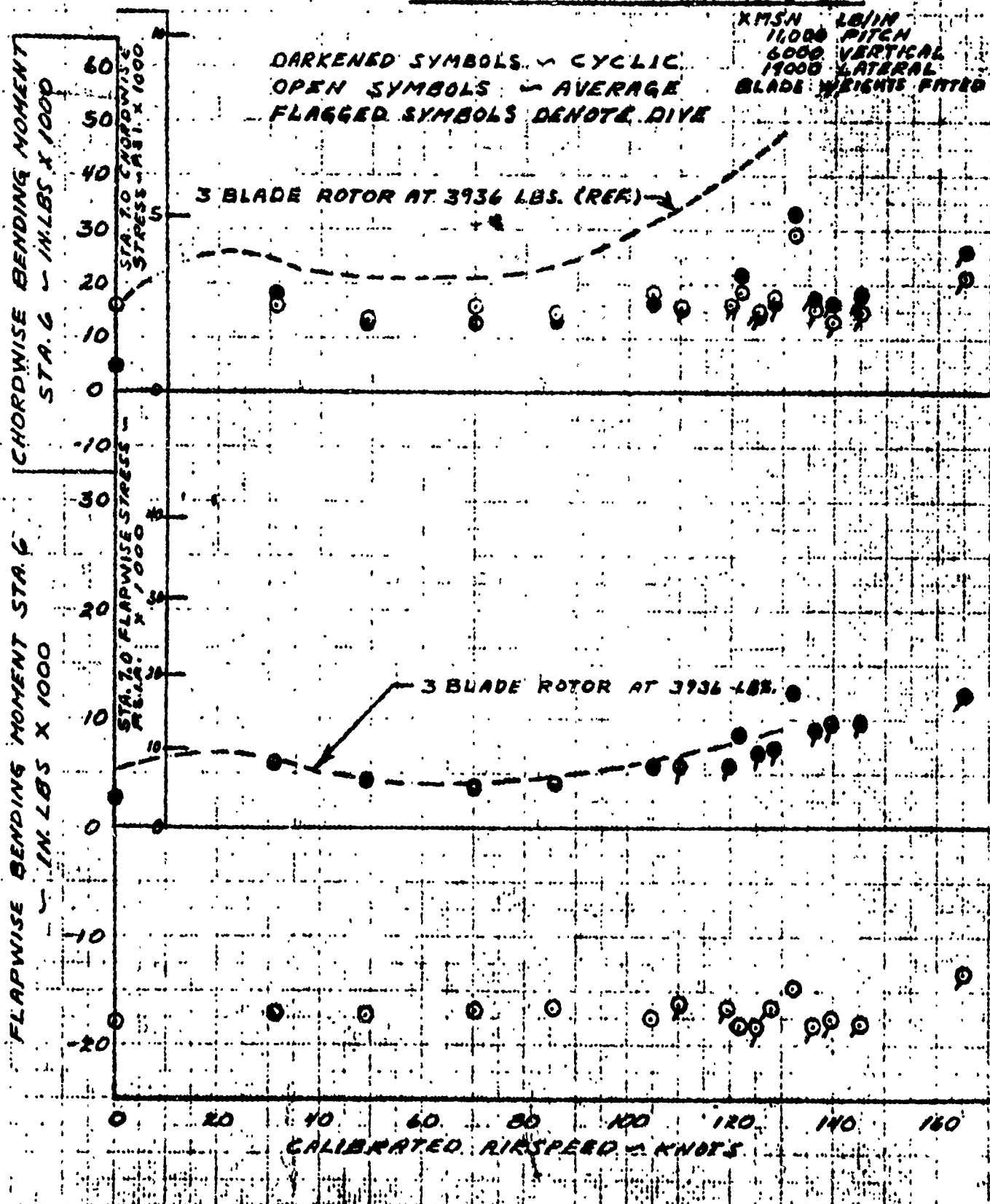
MOMENT AT HUB STA. 6.0. TRANSMISSION SUSPENSION MODIFIED  
BLADE WEIGHTS INCORPORATED

SF. 51.1  
FIG 4

# MAIN ROTOR BLADE LOADS VS. CALIBRATED AIRSPEED

TEST 373 F 385 4 BLADE ROTOR

T.O. WT. - 4006 HORIZONTAL STABILIZER - 3 DEG.



FORM 8879A

MOMENT AT HUB STA. 6.0 FINAL PHASE II CONFIG. TAIL PLANE - 3°

S.F. 62.1  
FIG 5



STA. 7.0 FLAPWISE STRESS PSI, Y1000

M.R. FLAPWISE BENDING MOMENT STA. 6.0 VS. LOAD FACTOR  
4 BLADE ROTOR

1.46 IN. FWD. C.G.

DARKENED SYMBOLS - CYCLIC MOM.  
OPEN SYMBOLS - AVERAGE MOM

X M S N

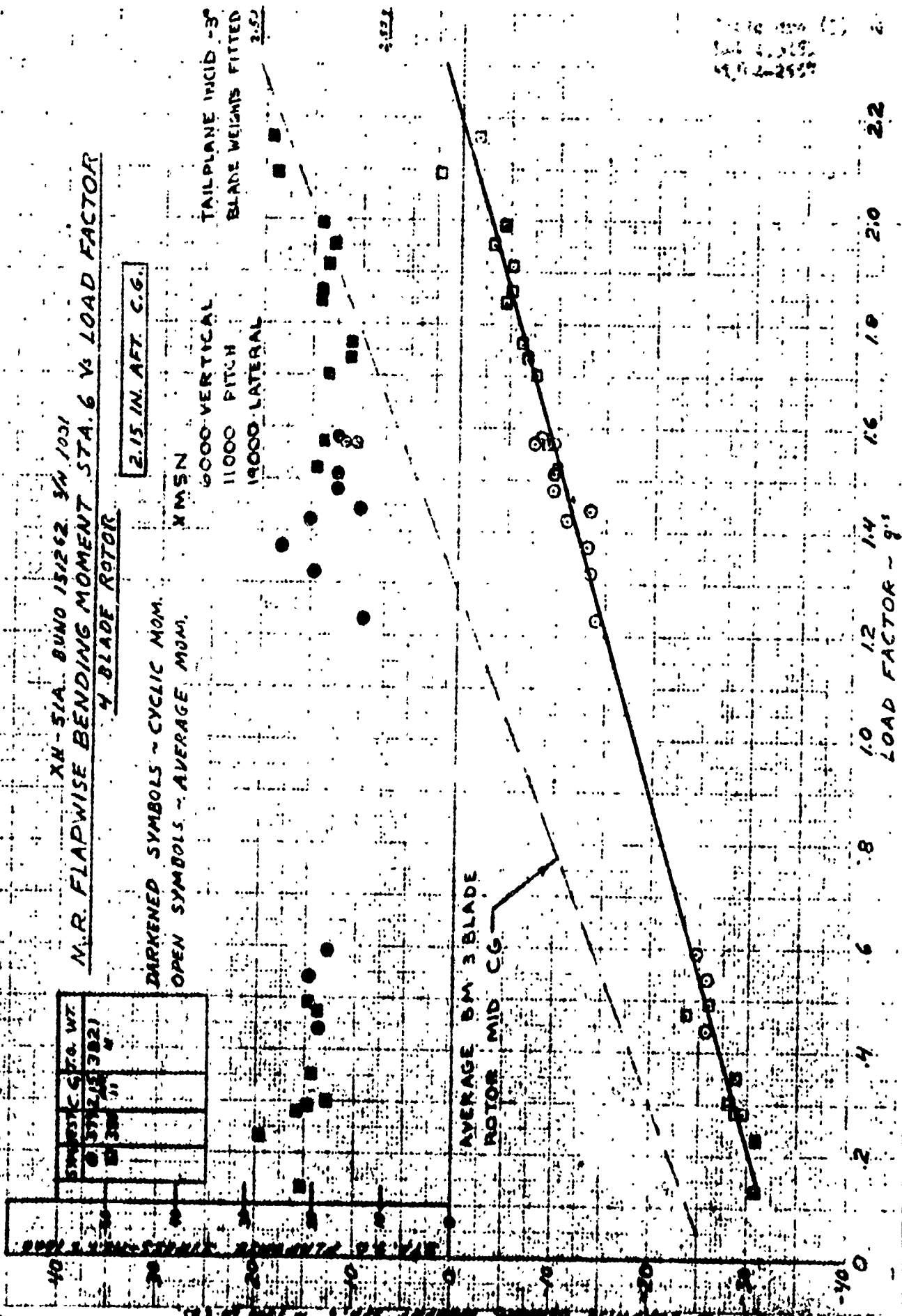
6000 VERTICAL  
11000 PITCH  
19000 LATERAL  
TAILPLANE INCID 3° - 51° ON TEST B71  
BLADE WEIGHTS FITTED

SYM	TEST	C.G.	T.G. WT.
Δ	371	1.46 FWD	4000
□	373	"	"
□	376	"	"
□	378	"	"

AVERAGE B.M. 3 BLADE  
ROTOR MID C.G.

LOAD FACTOR - 9" 14 16 18 20 22

FLAP B.M. AT STA. 6.0 VS. LOAD FACTOR C.G. 1.46" FWD. FINAL PHASE CONFIG. FIG 6



FLAP B.M. AT STA. 6.0 VS. LOAD FACTOR C.G. 2.15' AFT FINAL PHASE FIG 7

XM-51A BUNO 151262 SN 1001  
 M.R. CHORDWISE BENDING STA. 6 VS. LOAD FACTOR  
 4 BLADE ROTOR

1.46 W. FWD. C.G.

DARTED SYMBOLS - CYCLIC MOM.  
 OPEN SYMBOLS - AVERAGE MOM.  
 SMALL NUMBERS AT RIGHT OF  
 DARTED SYMBOLS ARE  
 INDICATED AIRSPEED.

XMSN  
 6000 VERTICAL  
 11000 PITCH  
 19000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID - 3°  
 - 5.1° ON TEST 371

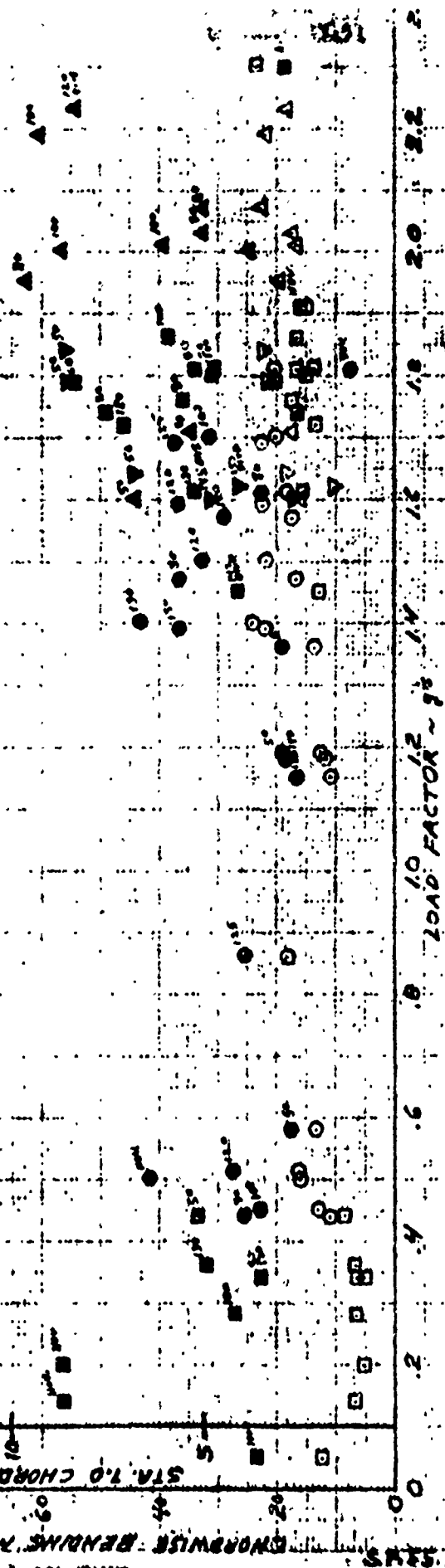
SYM	TEST	C.G.	TO WT.
△	371	1.46 FWD	4006
□	373	"	"
□	376	"	"
□	378	"	"

STA. 1.0 CHORDWISE STRESS P.S.I. X 1000

CHORDWISE BENDING MOMENT STA. 6 - 1000 IN. LB.

CHORD B.M. AT HUB STA. 6.0 VS. LOAD FACTOR C.G. 1.46" FWD  
 FINAL PHASE II CONFIG.

FIG 8



XN-51A BUHO 151262 3/4/1201  
 M.R. CHORDWISE BENDING STA. 6 VS. LOAD FACTOR  
 4 BLADE ROTOR

2.15 IN. AFT C.G.

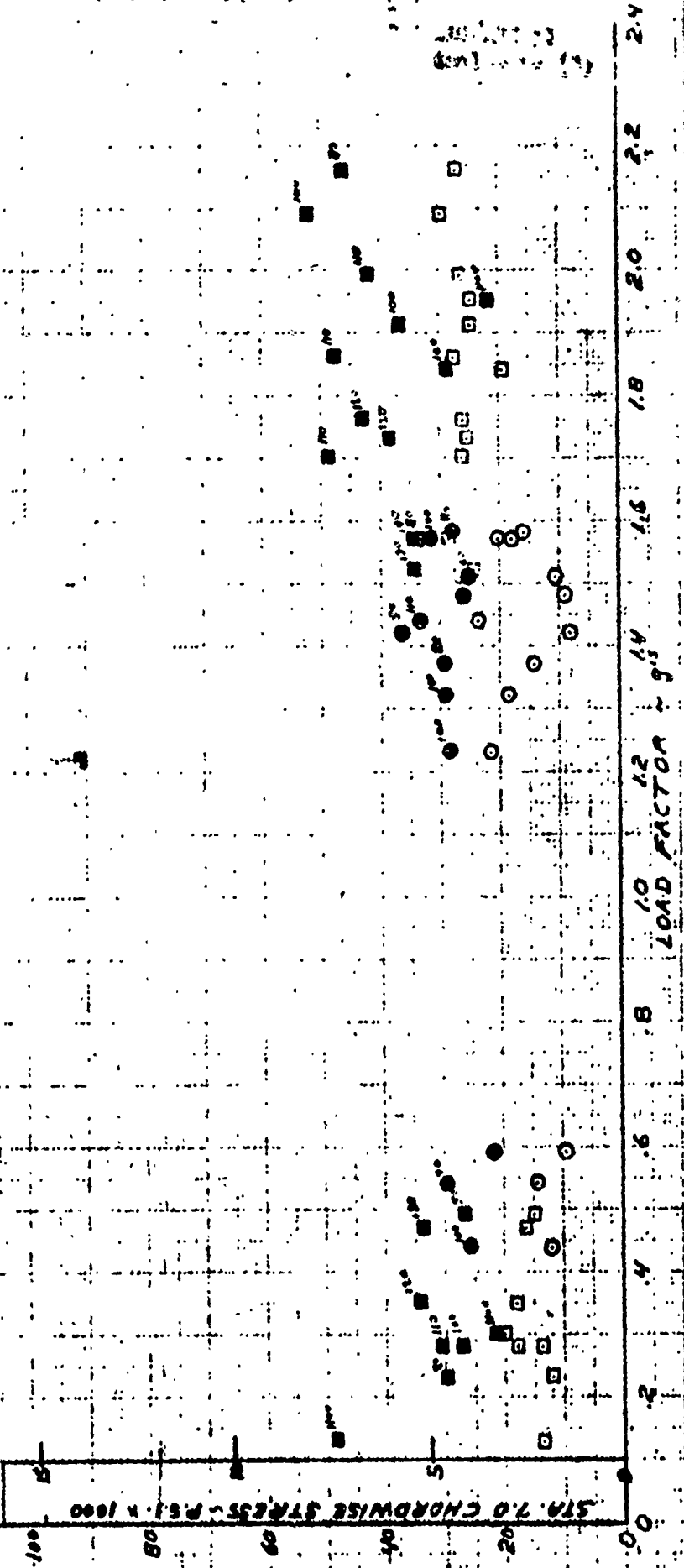
DARKENED SYMBOLS - CYCLIC MOM.  
 OPEN SYMBOLS - AVERAGE MOM.  
 SMALL NUMBERS AT RIGHT OF  
 DARKENED SYMBOLS ARE  
 INDICATED AIR SPEED

XMSN 16/IN  
 6000 VERTICAL  
 11000 PITCH  
 11000 LATERAL  
 BLADE WEIGHTS FITTED  
 TAILPLANE INCID - 3°

SYM TEST	C.G.	TR. WT.
○ 379	2.15 AFT	3821
□ 380	"	"

CHORD B.M. AT HUB STA. 6.0 VS LOAD FACTOR C.G. 2.15" AFT FINAL PHASE II CONFIG  
 175.5 R.P.M.

STA 7.0 CHORDWISE STRESS - PSI x 1000



# XN-51A BUND 151262 3/4/1961 M.R. FLAPWISE BENDING MOMENT VS. ROTOR SPAN

4 BLADE ROTOR - HIGH SPEED LEVEL

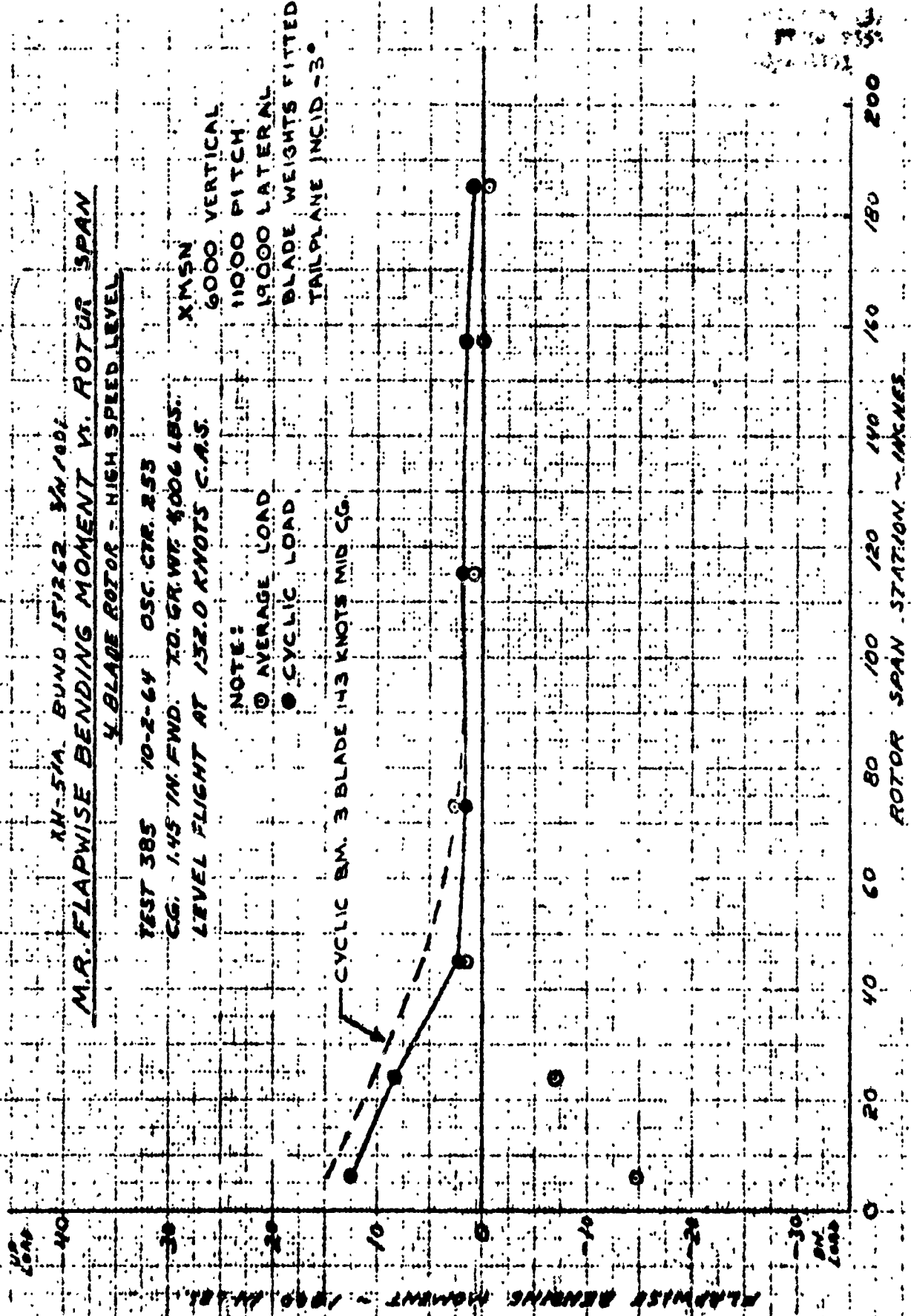
TEST 385 10-2-64 OSC CTR. 253  
CG. 1.45 IN. FWD. TO GR.WT. 4006 LBS.  
LEVEL FLIGHT AT 132.0 KNOTS C.A.S.

XMSN  
6000 VERTICAL  
11000 PITCH  
19000 LATERAL  
BLADE WEIGHTS FITTED  
TAILPLANE INCID -3°

NOTE:

○ AVERAGE LOAD  
● CYCLIC LOAD

CYCLIC BM. 3 BLADE 143 KNOTS MID CG.



FORM 5174A

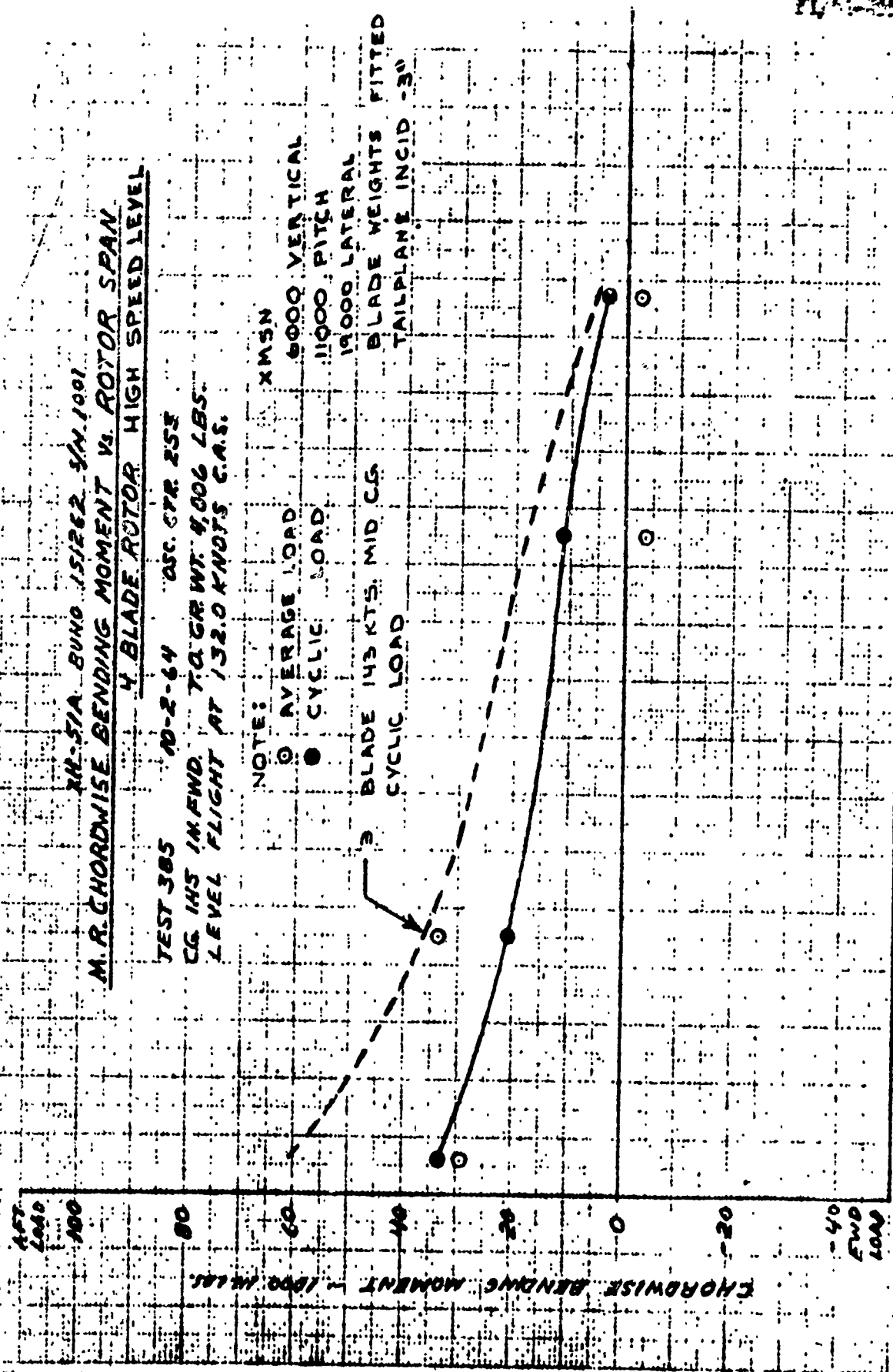
FLAPBENDING MOMENT VS. SPAN-LEVEL FLT. FINAL PHASE III CONFIG.

S.F.62.1  
FIG 10

See 14-5008  
 26 JAN/43/172  
 FL 1-1-50

**M.R. CHORDWISE BENDING MOMENT VS. ROTOR SPAN**  
**4 BLADE ROTOR HIGH SPEED LEVEL**

TEST 385 10-2-64 OSC. CTR. 253  
 CG 145 IN FWD. T.O. GRWT. 4,006 LBS.  
 LEVEL FLIGHT AT 132.0 KNOTS C.A.S.



NOTE: XMSN

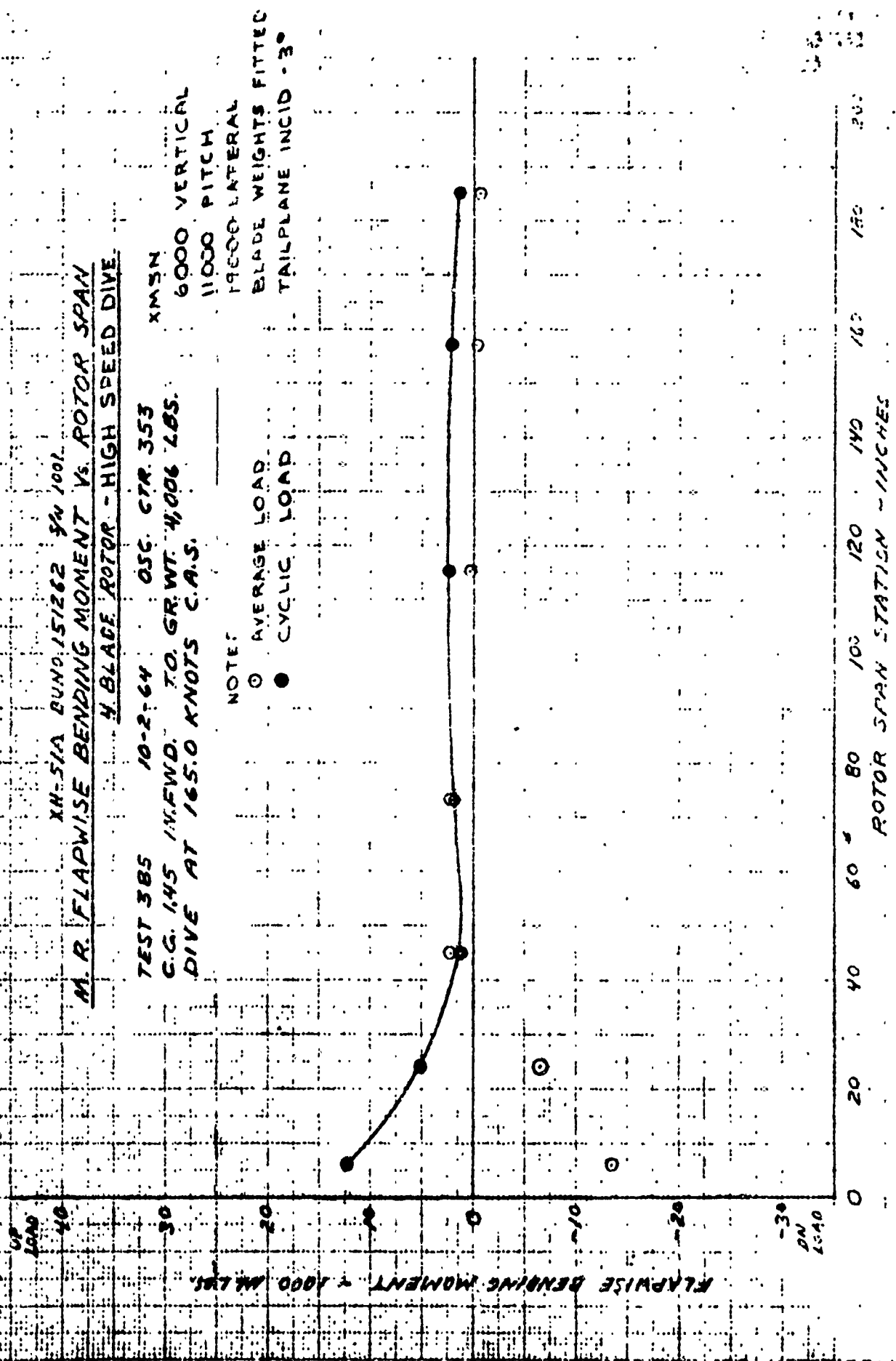
○ AVERAGE LOAD 6000 VERTICAL  
 ● CYCLIC LOAD 11000 PITCH  
 3 BLADE 143 KTS. MID CG 19000 LATERAL  
 CYCLIC LOAD BLADE HEIGHTS FITTED  
 TAILPLANE INCID -30

CHORD BEND. MOM. VS. SPAN LEVEL FLT. FINAL PHASE 22 CONFIG.

S.F. 13.1  
 FIG 11

FROM DATA

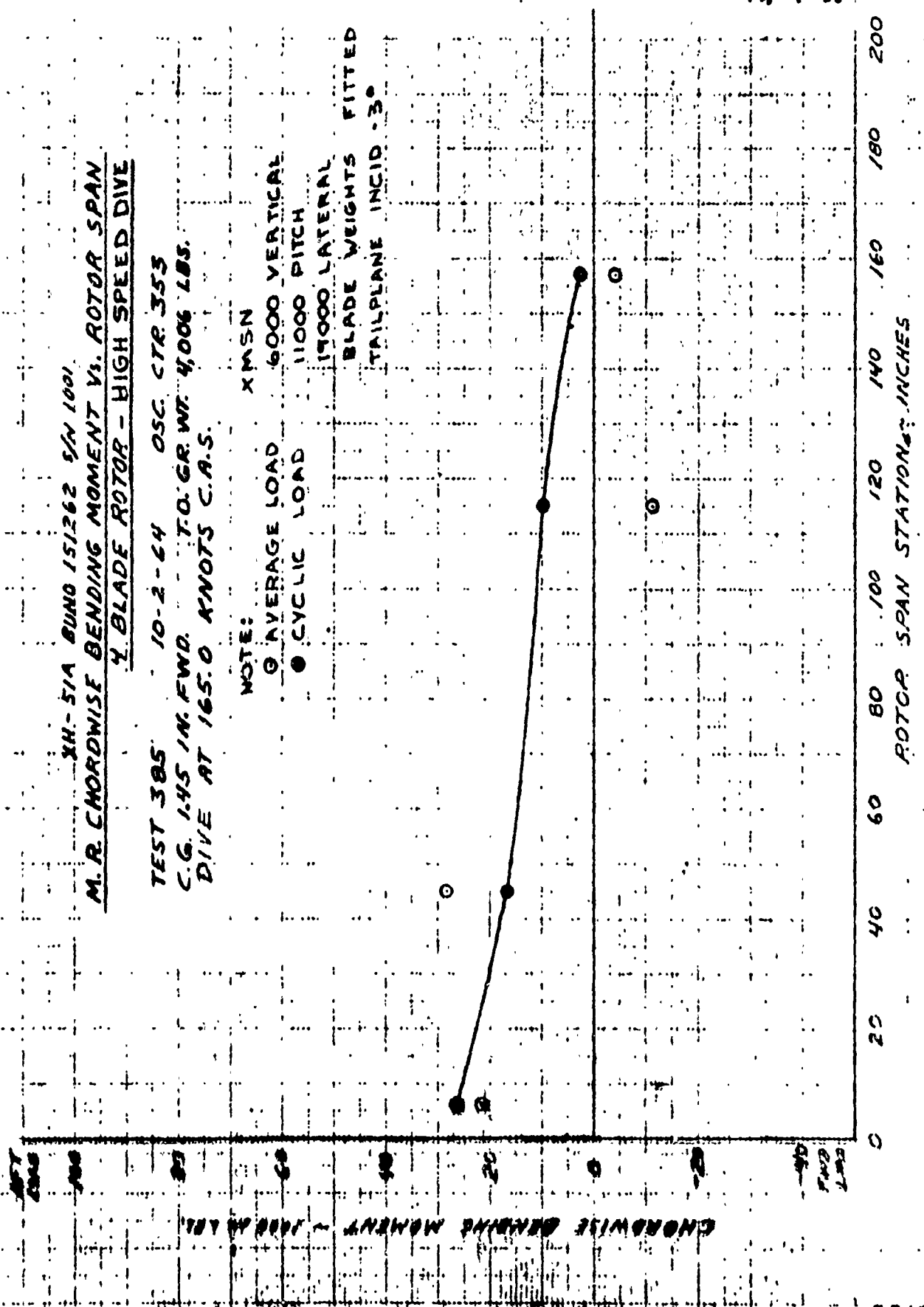
FLAP BENDING MOMENT VS. SPAN IN DIVE FINAL PHASE III CONFIG.



XH-51A BUONO 151262 SN 1001  
M.R. CHORDWISE BENDING MOMENT VS. ROTOR SPAN  
4 BLADE ROTOR - HIGH SPEED DIVE

TEST 385 10-2-64 OSC. CTR 353  
 C.G. 1.45 IN. FWD. T.O. GR. WT. 4006 LBS.  
 DIVE AT 165.0 KNOTS C.A.S.

NOTE: XMSN  
 ○ AVERAGE LOAD 6000 VERTICAL  
 ● CYCLIC LOAD 11000 PITCH  
 17000 LATERAL  
 BLADE WEIGHTS FITTED  
 TRAILPLANE INCID. 3°



CHORD BENDING MOMENT VS. SPAN IN DIVE FINAL PHASE II CONFIG. SP. 65.1  
 FIG 13



# XH-51A BUONO 151262 SIN 1001 MR. FLAPWISE BENDING MOMENT VS. ROTOR SPAN

## 4 BLADE ROTOR - HIGH 'G' TURN IN DESCENT

TEST 376 9-23-64 OSC. CTR. 564  
CG. 146 IN. FWD. GR. WT. 3,965 LBS.  
DESCENT AT 118.5 KNOTS C.A.S.  
2.23 G'S AT 3,965 LBS.  
2.53 G'S CORRECTED TO 3,500 LBS.

XMSN

6000 VERTICAL

11000 PITCH

19000 LATERAL

BLADE WEIGHTS FITTED

TAILPLANE INCID -3°

NOTE: ○ AVERAGE LOAD  
● CYCLIC LOAD

UP  
LOAD

FLAPWISE BENDING MOMENT - 1000 IN. LB.

DOWN  
LOAD

0 20 40 60 80 100 120 140 160 180 200  
ROTOR SPAN STATION - INCHES

FLAP BENDING MUMENT VS. SPAN IN MANEUVER FINAL PHASE II CONFIG.

S.F.57.  
FIG 1

AFT  
LOAD

100

80

60

40

20

0

-20

-40

DN.  
LOAD

CHORDWISE BENDING MOMENT - 1000 IN. LBS.

FORM 42-1A

CHORD BENDING MOM. VS. SPAN IN MANEUVER FINAL PHASE II CONFIG.

S.F. 58.1  
FIG 15

XH-51A BUNO 151212 SIN 1001

M.R. CHORDWISE BENDING MOMENT VS. ROTOR SPAN

4 BLADE ROTOR HIGH G TURN IN DESCENT

TEST 376 9-23-64 OSC. CTR. 564

G.G. 1.16 IN. FWD. GR.WT. 3,965 LBS.

DESCENT AT 118.5 KNOTS C.A.S.

2.23 G'S AT 3,965 LBS.

2.53 G'S CORRECTED TO 3500 LBS.

XM SN

6000 VERTICAL

11000 PITCH

19000 LATERAL

BLADE WEIGHTS

FITTED

TAIL PLANE INCID -3°

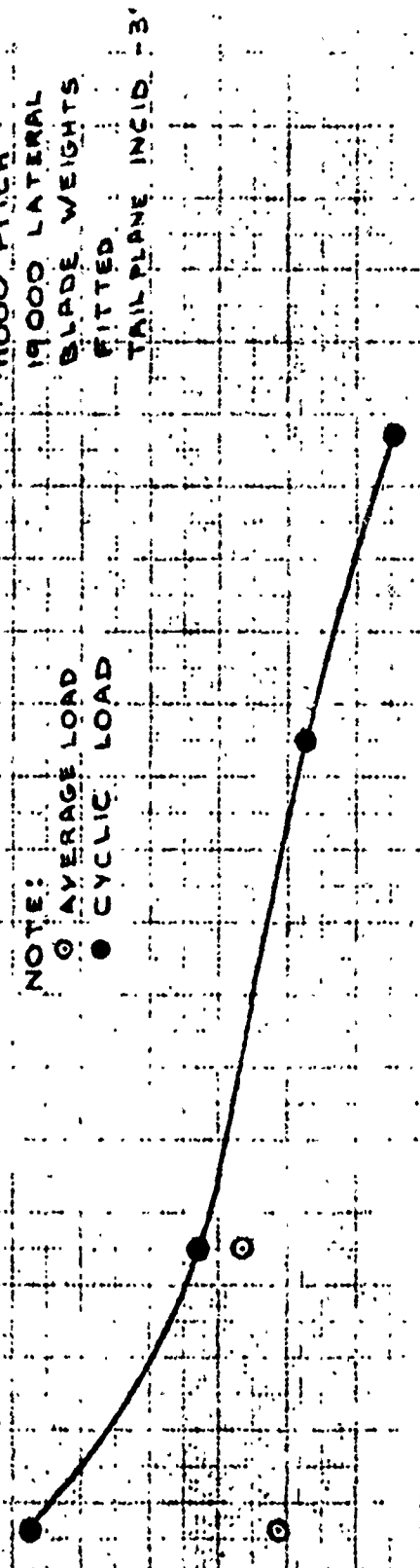
NOTE:

○ AVERAGE LOAD

● CYCLIC LOAD

0 20 40 60 80 100 120 140 160 180 200

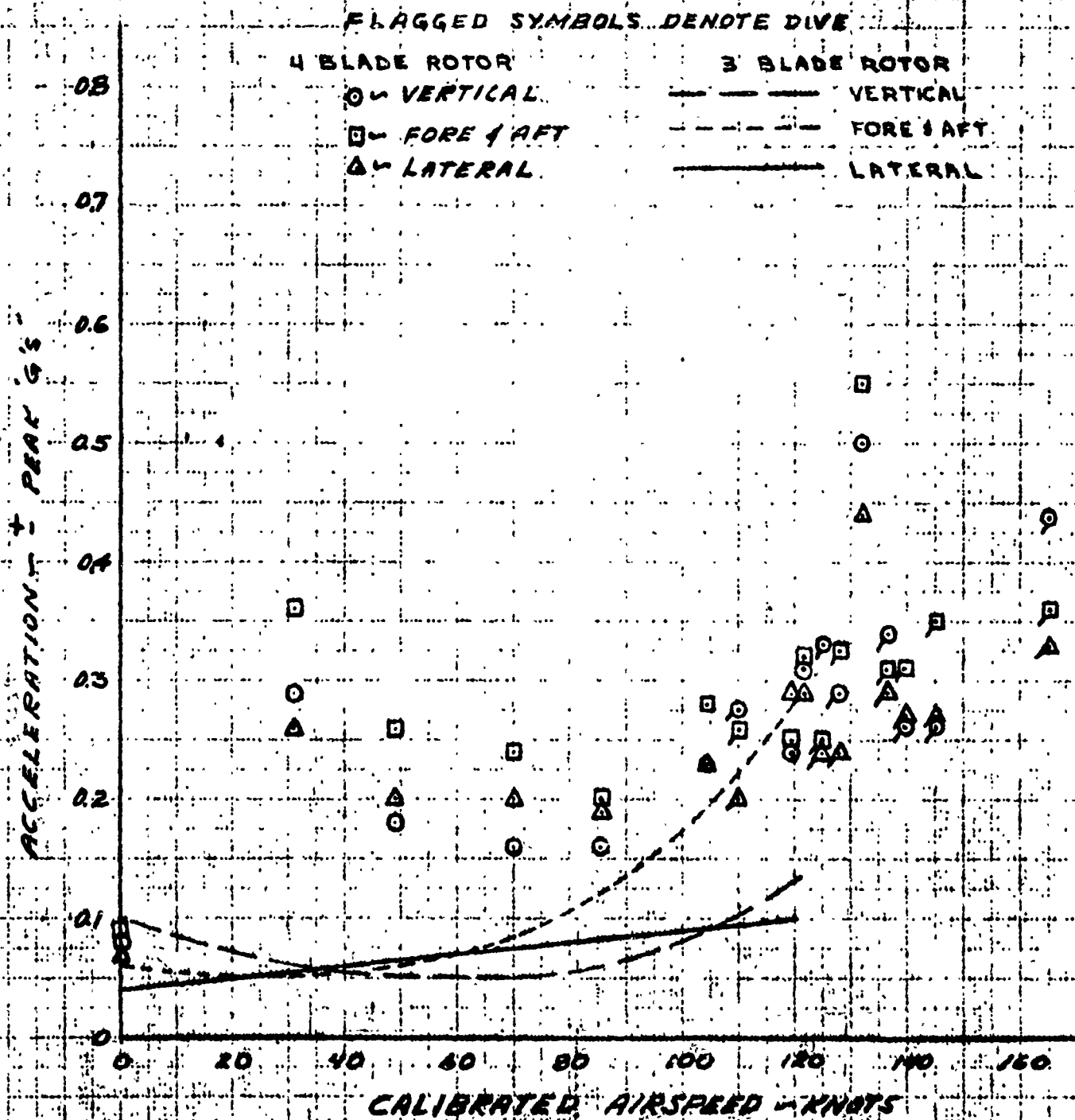
ROTOR SPAN STATION - INCHES



# CABIN VIBRATIONS VS. CALIBRATED AIRSPEED

TEST 373 1385

4 PER REV.



FORM 8870A

CABIN VIBRATION LEVEL

FINAL PHASE & CONFIG.

5.559.1

FIG 16

# CYCLIC CONTROL POSITIONS IN LEVEL FLIGHT

4-BLADE ROTOR SYSTEM

FWD C.G. LOCATION

SHIP: BUNO 151262

CYCLIC STICK ROLL POSITION-IN

RIGHT

2

1

0

1

2

LEFT

APR

1

0

1

2

3

4

5

6

FWD 20

40

60

80

100

120

140

TRUE AIRSPEED - KNOTS

## AVERAGE TEST CONDITIONS

SYM	TEST	WAVE LEN	DEN REF	RPM	LONG. MOM.	LAT. MOM.
0	384	3910	1250	22.5	-61.0 IN-LB FT.	+755 IN-LB FT.

TRAVEL LIMIT: 2.81 IN. RIGHT-2.81 IN. LEFT.

3-BLADE ROTOR

## CONFIGURATION NOTES:

1. CYCLIC STICK PITCH SENSITIVITY = 100%.
2. LANDING GEAR UP.
3. SPEED SENSOR OFF.

CYCLIC STICK PITCH POSITION-IN

3-BLADE ROTOR 100% SENSITIVITY

APR TRAVEL LIMIT = 4.75 IN.

FWD TRAVEL LIMIT = 6.125 IN.

CONTROL TO TRIM IN LEVEL FLIGHT

FIGURE 17

# STATIC LONGITUDINAL STABILITY

FWD C.G. LOCATION

FOUR BLADE ROTOR SYSTEM

SNIP: BUW 151262

CYCLIC STICK PITCH FORCE IN LB  
PULL  
10  
5  
0  
5  
10  
PUSH

TEST	WAVE NO.	DEV. ASMT	LONG. MON.	LAT. MON.
386	3588	4200	-3390 WAVE FWD 14-18 IN	7317 14-18 IN

○△ - TRIM

○△ - RETURN

## CONFIGURATION NOTES:

1. CYCLIC STICK PITCH SENSITIVITY = 100%
2. LANDING GEAR UP.
3. SPEED SENSOR OFF.

AFT  
1  
0  
1  
2  
3  
4  
5  
6  
CYCLIC STICK PITCH POSITION IN.  
FWD 20 40 60 80 100 120 140  
CALIBRATED AIRSPEED - KNOTS

AFT TRAVEL LIMIT = 4.75 IN.

FWD TRAVEL LIMIT = 6.125 IN.

CONSTANT COLLECTIVE STATIC LONGITUDINAL STABILITY

(1000)

FIGURE 1B

FORM 857C

993A

LOCKHEED HELICOPTER

Model XH-51A

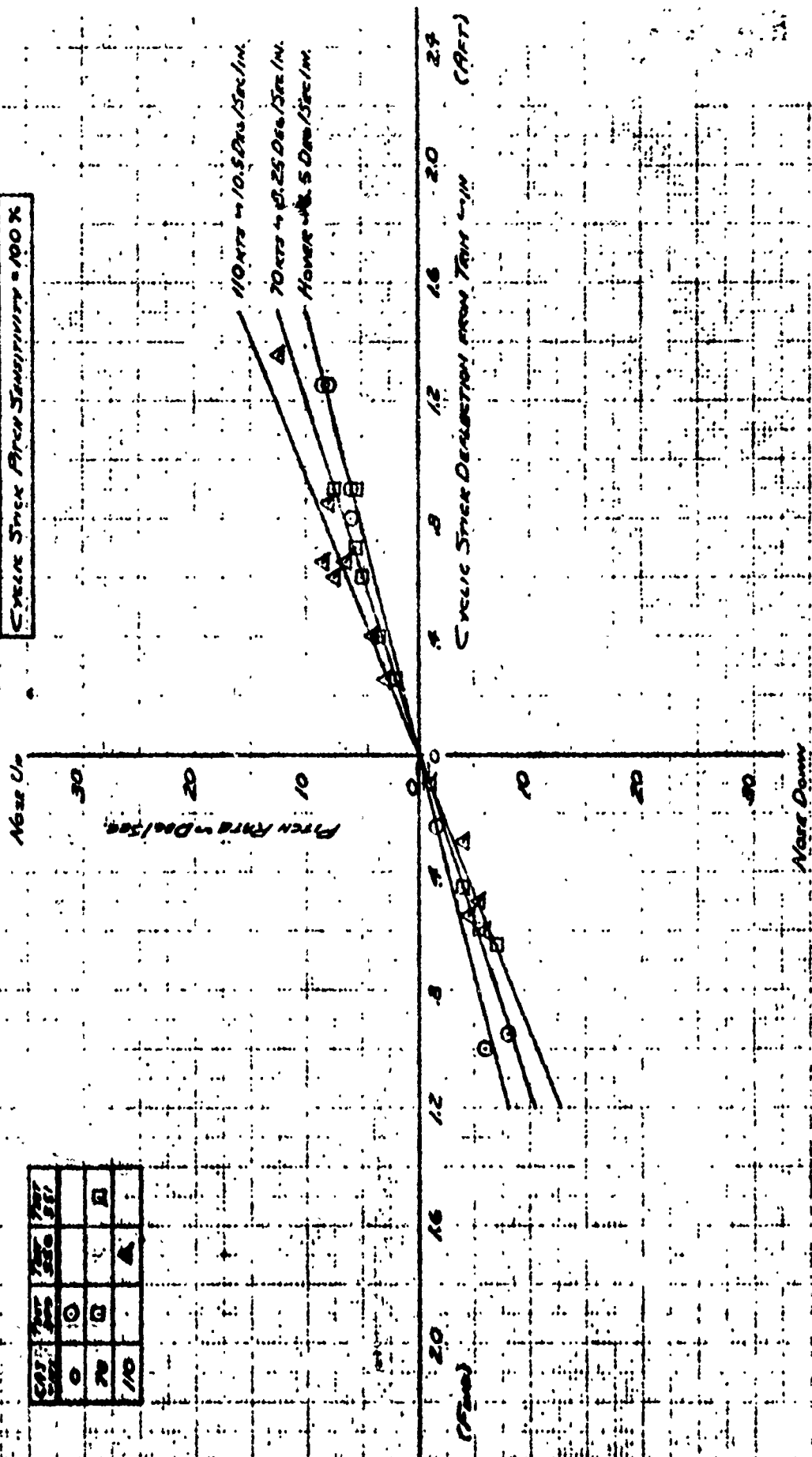
# LONGITUDINAL CONTROL POWER

Four Blade Main Rotor

SNIP: 6810151262

Cyclic Stick Pitch Sensitivity = 100 %

Calc	YPR	YPR	YPR	YPR
Stick	Stick	Stick	Stick	Stick
0	0	0	0	0
20	0	0	0	0
10	0	0	0	0



FORM 8876A

LONGITUDINAL CONTROL POWER

FIGURE 20

(1001)

979A

LOCKHEED HELICOPTER  
Model XH-51A

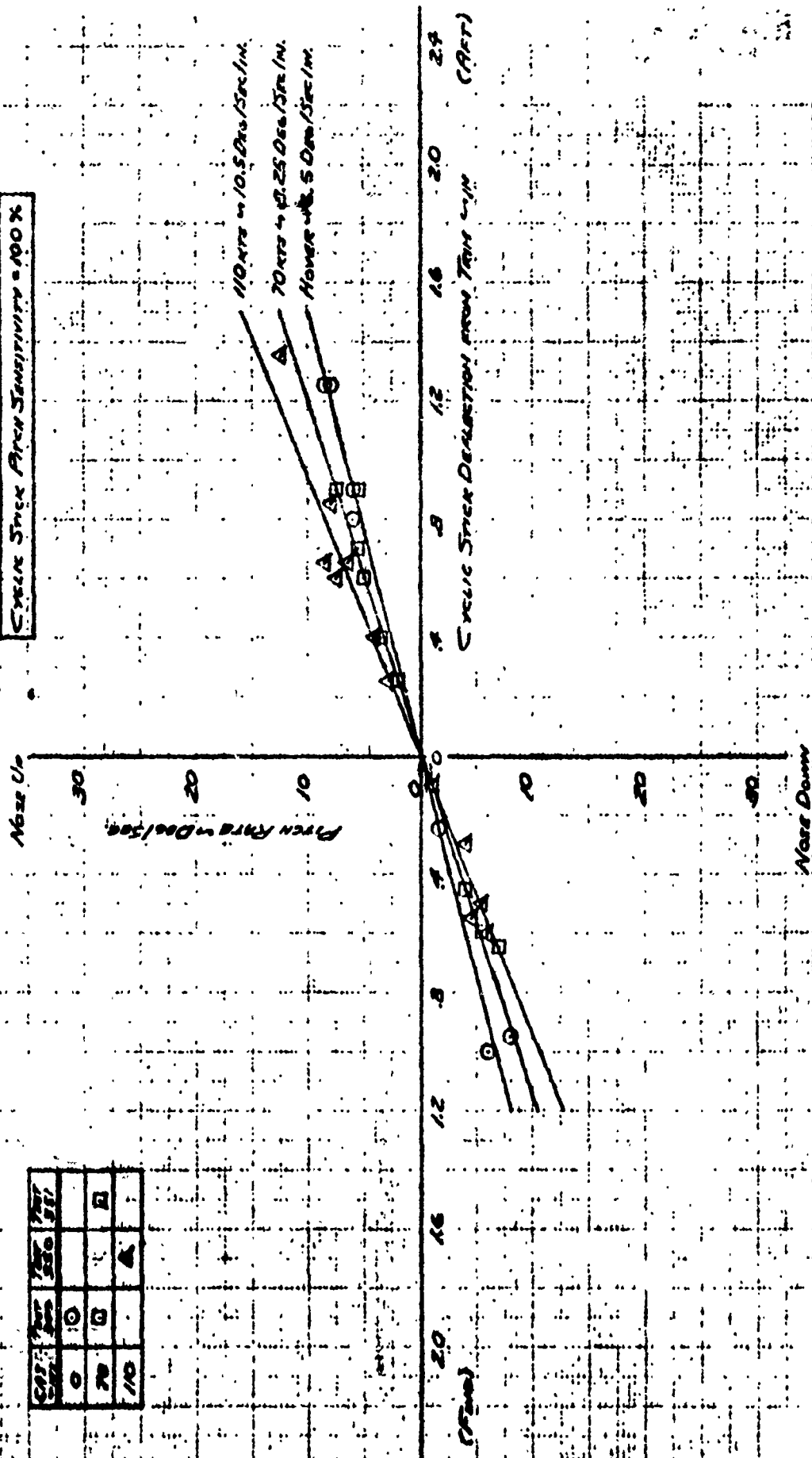
# LONGITUDINAL CONTROL POWER

Four Blade Main Rotor

SNIP: 6040 151262

Cyclic Stick Pitch Sensitivity = 100%

CAT	THS	THS	THS
0	10	10	10
75	10	10	10
100	10	10	10



LONGITUDINAL CONTROL POWER

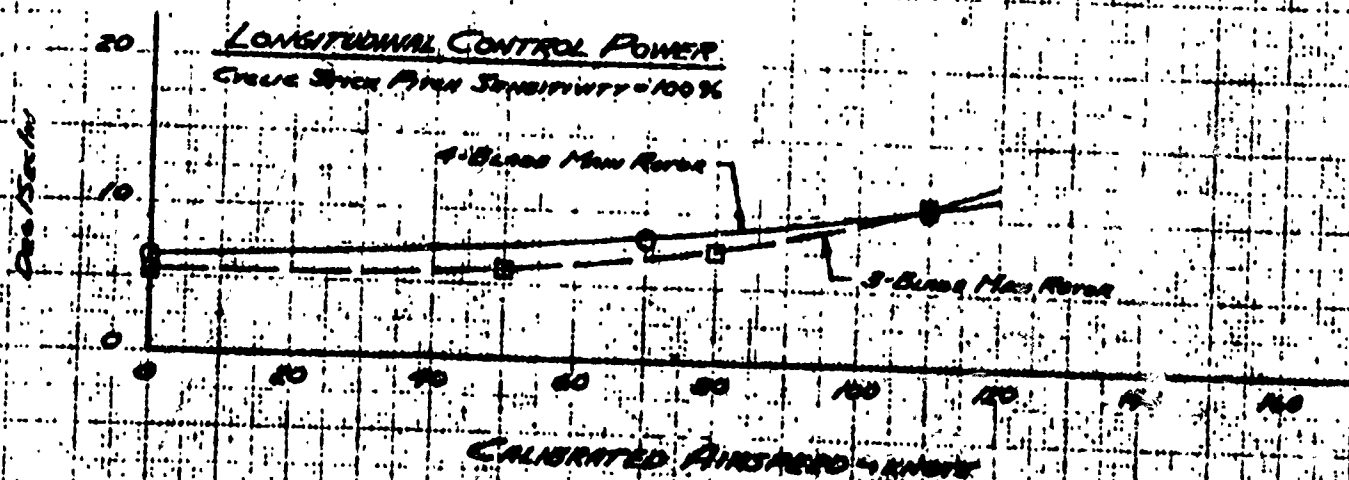
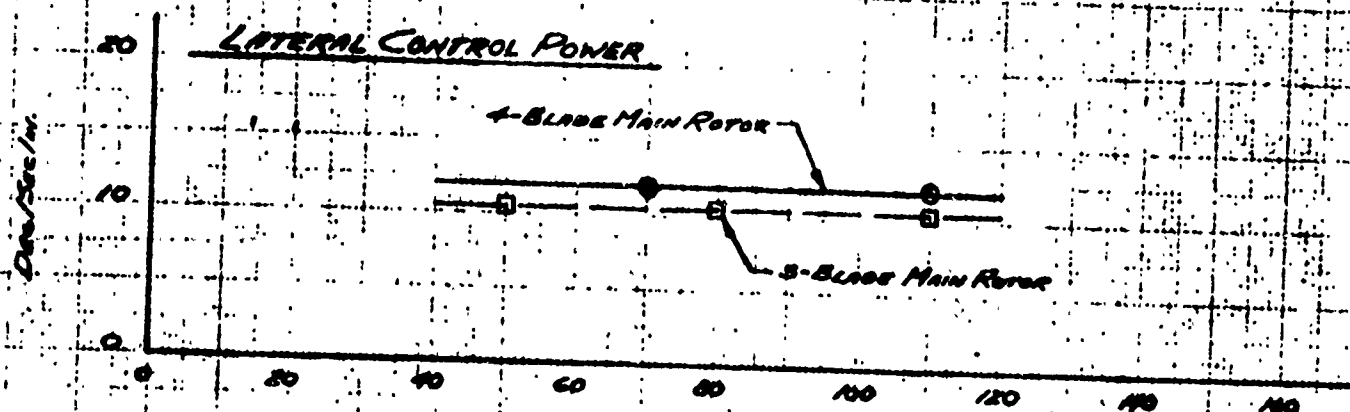
FIGURE 20

(100)

979A

# CONTROL POWER COMPARISON BETWEEN THE 3 AND 4 BLADE ROTOR SYSTEMS

SN: BND 15828



CONTROL POWER COMPARISON - 3 BLADE & 4 BLADE ROTORS

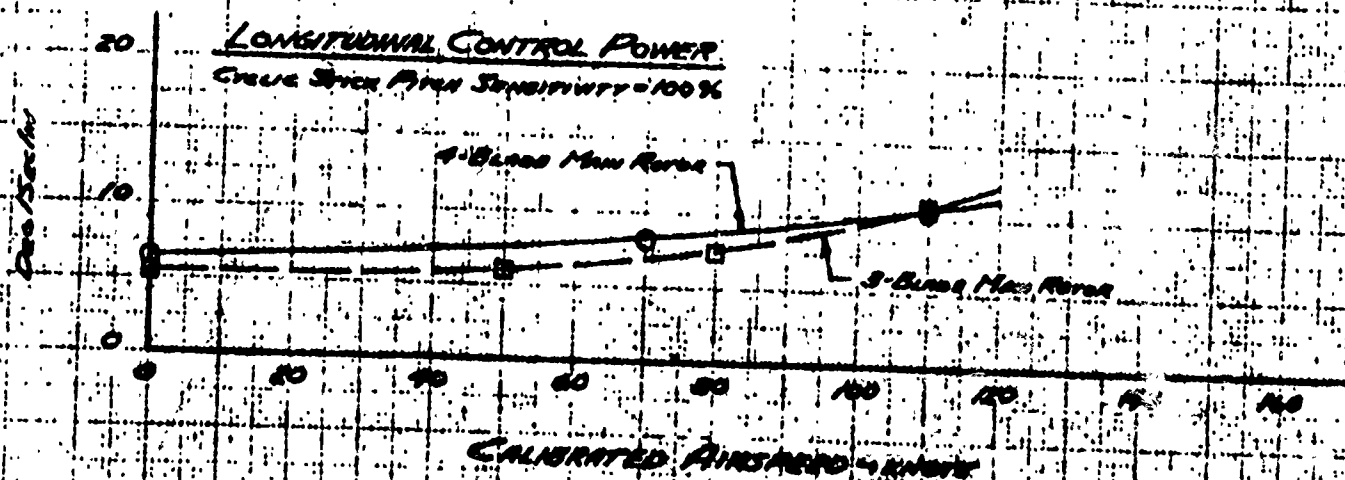
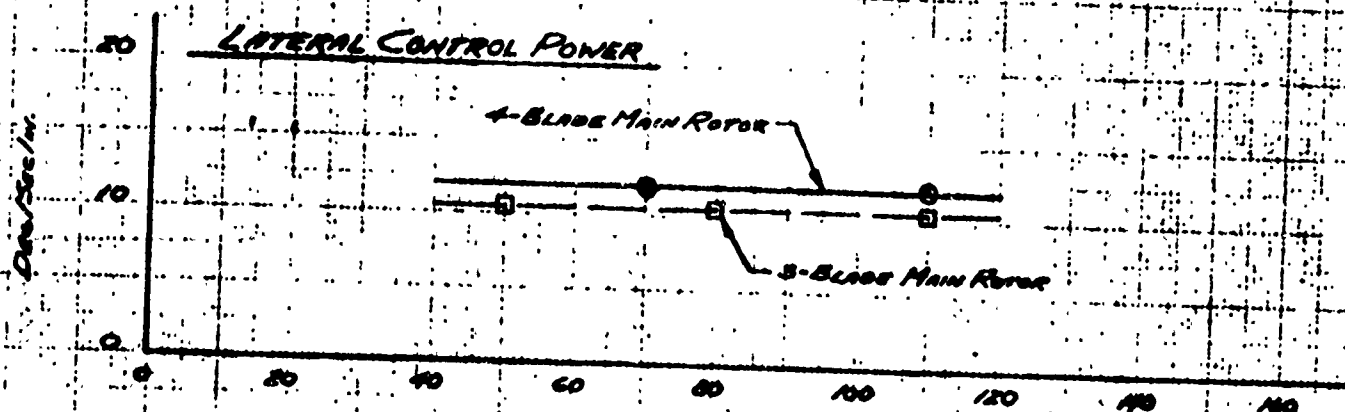
FIGURE 22

2809



# CONTROL POWER COMPARISON BETWEEN THE 3 AND 4 BLADE ROTOR SYSTEMS

SHW: BMD 15182



FORM 8570A

CONTROL POWER COMPARISON - 3 BLADE & 4 BLADE ROTORS

FIGURE 22

280A

# MANEUVERING STABILITY

FWD C.G. LOCATION

FOUR BLADE ROTOR SYSTEM

SHAW-BUNO 151262

## CONFIGURATION NOTES:

1. CYCLE STICK PITCH SENSITIVITY = 100%.
2. EXTERNAL TUNING WEIGHTS ON MAIN ROTOR BLADES.
3. LANDING GEAR UP.
4. SPEED SENSOR OFF.
5. 31.5 LB BOE-WRIGHT INSTALLED.

CYCLE STICK PITCH F. REE VLS

TEST	FLY	WIND	DEN	LONG.	LAT.
386	244	3388	1300	-8530	1517

LONG FACTOR  $\times 10^5$

MANEUVERING STABILITY 4 BLADE SYSTEM

FIGURE 23 994A

# MANEUVERING STABILITY

MID C.G. LOCATION

SHIP: BUNO 151262

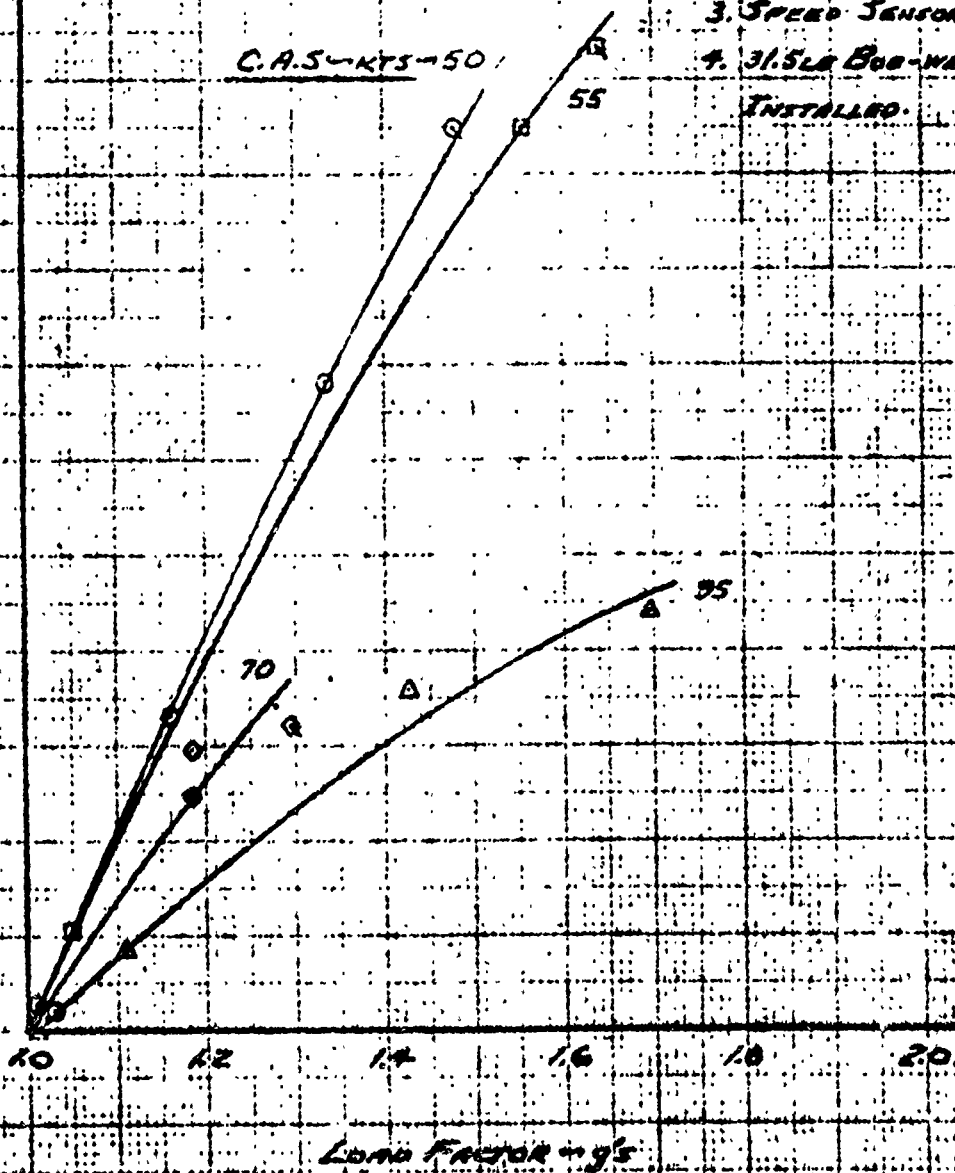
THREE BLADE ROTOR SYSTEM

SYM	TEST	WAVE H/L	GEN ALT-Ft	LONG MAN.	LAT. MAN.
① Δ ②	279	3710	5600	0	2-57 IN-18-0
③ Δ ④	303	3630	2350	588 W-18-0	2109 -47

## CONFIGURATION NOTES:

1. CYCLIC STICK PITCH  
SENSITIVITY = 100%.
2. LANDING GEAR RETRACTED
3. SPEED SENSOR OFF.
4. 31.5 LB BOB-WEIGHT  
INSTALLED.

CYCLIC STICK PITCH FORCE IN LB



FORM 4878

MANEUVERING STABILITY 3 BLADE ROTOR  
FIGURE 24

879A  
(1001)

# MANEUVERING STABILITY

FWD. C.G. LOCATION

SHIP: BUNO 151262

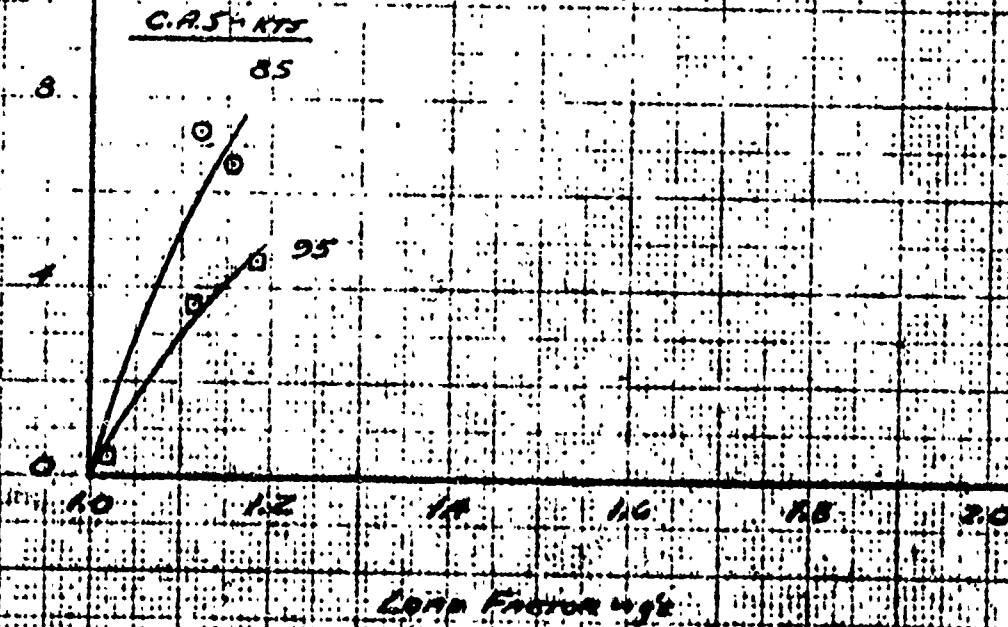
THREE BLADE ROTOR SYSTEM

CYCLIC STICK PITCH FORCE - LB

SYM	TEST	WAVE NO.	DRN. ALTITUDE	LONG. MOM.	LAT. MOM.
0 0	316	3890	7900	5115 IN-LB	727 IN-LB

## CONFIGURATION NOTES:

1. CYCLIC STICK PITCH SENSITIVITY = 100%.
2. LANDING GEAR RETRACTED.
3. SPEED SENSOR OFF.
4. 315 LB BOB-WEIGHT INSTALLED



FORM 5278

MANEUVERING STABILITY 3 BLADE ROTOR  
FIGURE 25

80/1  
(100)

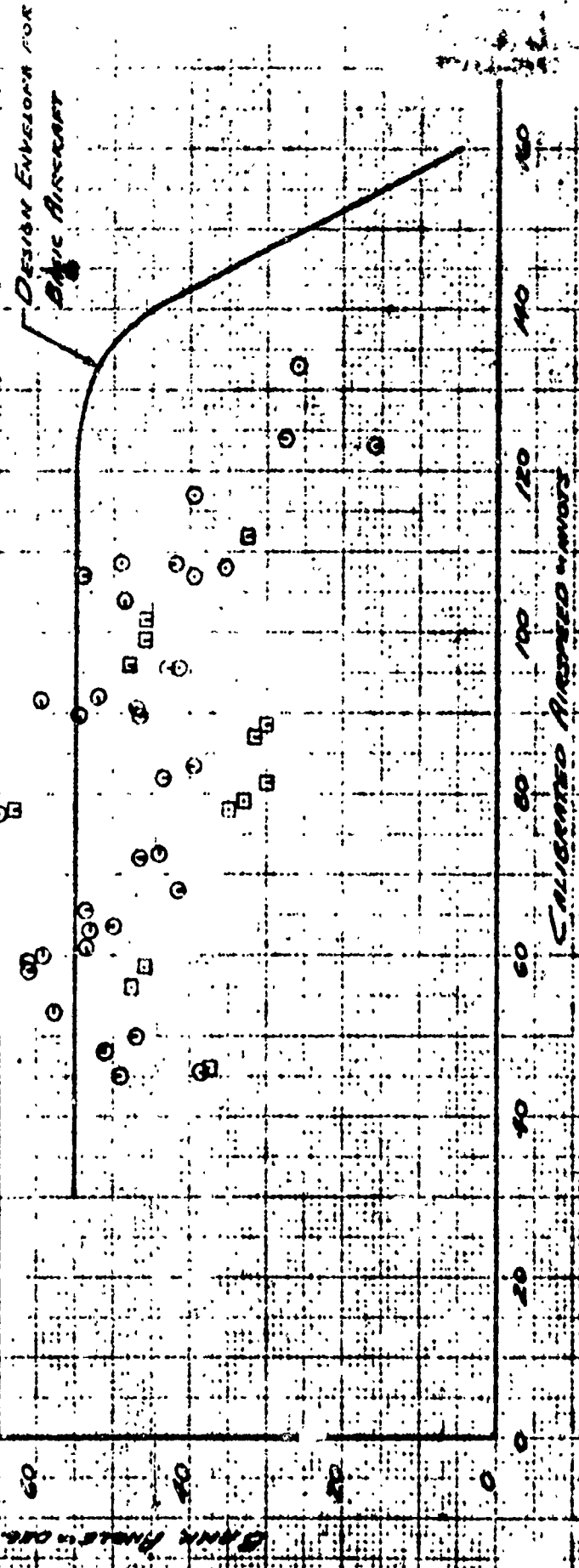
LOCKHEED HELICOPTER  
MODEL XH-5HA

# BANK ANGLE - VELOCITY ENVELOPE

SHIP: BUONO 151262  
FWD AND AFT C.G. LOCATION  
FOUR BLADE MAIN ROTOR

SIN	TEST	Wing W/B	Low Man. W/B	LAR Man. W/B
①	371	3965	3680	3778
②	372	3975	3690	3778
③	376	3998	3720	3778
④	378	3975	3700	3778
⑤	386	3988	3850	3817

SIN	TEST	Wing W/B	Low Man. W/B	LAR Man. W/B
①	379	3765	3730	3778
②	380	3785	3750	3778



ANGLE OF BANK VELOCITY ENVELOPE

(100/1)

# LOAD FACTOR REQUIRED TO MAINTAIN A GIVEN ROTOR RPM IN AUTOROTATION

SHIP: BUNO 151262

TEST 383, FLIGHT 241

## NOTES:

1. 4 BLADE MAIN ROTOR WITH EXTERNAL  
TUNING WEIGHTS AT THE 5-FT RADIUS.

2. LANDING GEAR RETRACTED.

3. ENTRY SPEED = 76±2 KTS, C.A.S.

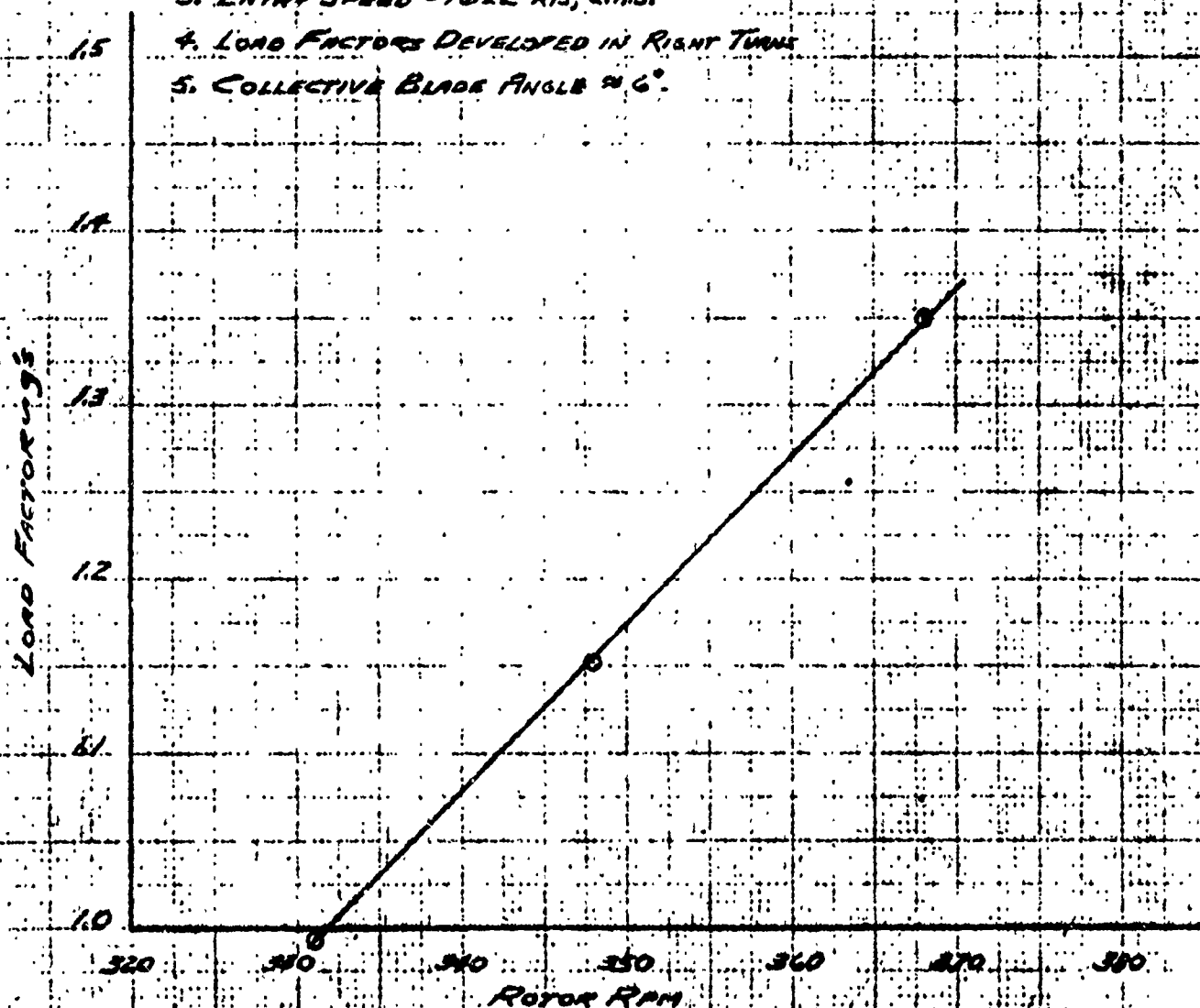
4. LOAD FACTORS DEVELOPED IN RIGHT TURN

5. COLLECTIVE BLADE ANGLE 26°.

W=4000 LB

LONG. MOM = 1720 IN-LB FWD

LAT. MOM = 198 IN-LB TO R.



LOAD FACTOR - ROTOR SPEED IN AUTOROTATION

FIGURE 27

(100)

986

# FIGURE 20

## COMPARISON OF 3-BLADE AND 4-BLADE ROTOR HOVER PERFORMANCE

SEA LEVEL STD. DAY 100% RPM

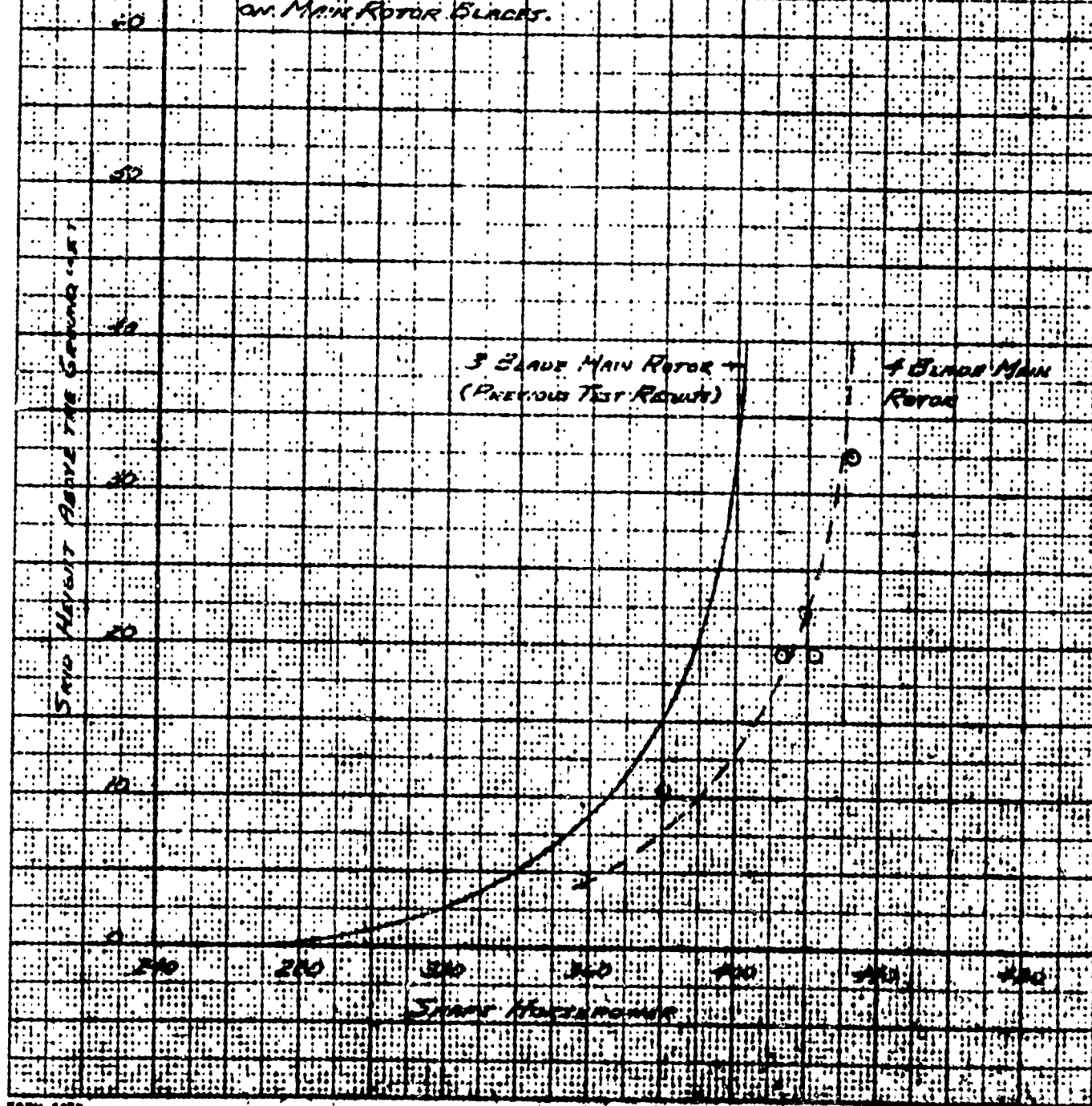
SMP: BUNO 151262

WGT 4050 LB

TEST 329

28 JULY 67

NO EXTERNAL TUNING WEIGHTS  
ON MAIN ROTOR BLADES.



HOVER PERFORMANCE

FIGURE 20

2547



LOCKHEED HELICOPTER  
MODEL XH-51A

# LEVEL FLIGHT PERFORMANCE

COMPARISON OF 3-BLADE AND 4-BLADE ROTOR SYSTEMS

SNIP: BUW 151262 Sea Level Standard Day - 100% RAN

LANDING GEAR RETRACTED

## Average Test Conditions

Test	Wing Area	Sea Level Altitude	RAN %	Engine Power	Engine RPM
1	384	3370	92.5	2760	1795
2	385	4000	93.0	2760	1817

4-Blade Main Rotor

W<sub>1/2</sub> = 4020 LB.

3-Blade Main Rotor

W<sub>1/2</sub> = 4130 LB.

5-Minute Horsepower

## Note:

THE 4-BLADE ROTOR SYSTEM USED DURING THIS TEST AND EXTERNAL TUNING WEIGHTS MOUNTED ON EACH BLADE AT THE 5.0 FT RADIUS. THE INCREASED PROFILE DUE TO THIS MODIFIED AIRFOIL HAS NOT BEEN DETERMINED.

TEST TRUE AIRSPEED - KNOTS